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RADIO/OPTICAL/STRAPDOWN INERTIAL GUIDANCE STUDY FOR ADVANCED KICK STAGE APPLICATIONS

Volume I - Summary

Prepared by
TRW, INC.
Redondo Beach, Calif.
for Electronics Research Center

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FOREWORD

This final report documents the results of the work accomplished under Tasks III and IV of a study of Radio/Optical/Strapdown Inertial Guidance Systems for future unmanned space missions, conducted by TRW Systems for the NASA Electronics Research Center under Contract NAS 12-141. This effort expands and extends the work accomplished previously under Tasks I and II of the same contract.

Volume I summarizes both the results of the study and recommendations reached, including those developed under Tasks I and II. Volume II documents the detailed study results for Tasks III and IV. (Volume II is in two parts. Part I is NASA CR-86197 and Part II is NASA CR-86198.)

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1. INTRODUCTION

This TRW Systems final report documents the results of a study of Radio/Optical/Strapdown Inertial Guidance Systems (ROI) for application to future unmanned space missions carried out under Contract NAS 12-141, Tasks III and IV. This effort extends and refines the work previously carried out and documented under Tasks I and II of the same contract (see Ref. 1-1). This volume presents a summary of results, conclusions, and recommendations for the entire study effort. Vol. II of this report presents the detailed study results for the work accomplished under Tasks III and IV.

Sec. 1 of this volume discusses the study objectives and constraints, the method of approach used in conducting the study, and the various assumptions made with regard to the mission definitions and vehicle characteristics related to the guidance and control system functions.

A summary of the principal conclusions and recommendations of the study are presented in sec. 2 which also includes a summary of conclusions and recommendations relative to the utilization of radio guidance for the missions under consideration.

A summary of the mission characteristics, guidance system operational sequences, and performance requirements is presented in sec. 3.

Sec. 4 summarizes the conceptual guidance and control system configurations developed to satisfy the mission and launch vehicle functional and interface requirements.

A summary of the results of the various performance analyses carried out to demonstrate the feasibility and to establish guidance system performance requirements is presented in sec. 5.

Sec. 6 summarizes the preliminary modular design of the radio/optical/strapdown inertial guidance and control system. Sec. 7 is a summary of the design and performance characteristics of the onboard optical and inertial sensing elements of the modular design.

1.1 STUDY OBJECTIVES

The primary objective of the study was to evaluate the feasibility of the "integrated modular design" concept for the guidance and control of launch vehicles and spacecraft for specified NASA unmanned space missions by means of analysis and design of a responsive system. Detailed study objectives were to:

- Establish the guidance and control requirements for a selected group of future NASA space missions.
- Investigate possible guidance concepts based on the appropriate use of radio, inertial, and optical techniques, with the further objective of establishing the functional role, capabilities, limitations, and constraints of each of these elements in the overall guidance system concept.
- Define feasible radio/optical/strapdown inertial navigation, guidance, and control system "conceptual designs."
- Perform analyses to establish the feasibility (performance) of the selected conceptual designs and to establish the significant performance characteristics of each component and subsystem.
- Perform a "preliminary modular design" of the radio/optical/strapdown inertial system meeting the composite requirements of all the missions considered, configured so that specific components may be interchangeably combined into given operational systems.
- Establish the performance capabilities of the preliminary modular design and verify by performance analyses that this design meets the performance requirements for each mission.
- Perform preliminary design studies of the inertial and electro-optical sensor subsystems and indicate areas of technology where state-of-the-art advances are necessary.

A "conceptual design" is a functional representation of the guidance and control system component configuration responsive to a given mission; it includes 1) a functional schematic blocking out each component subsystem, the mechanization of the various operational computations, all data flow, and all moding and switching functions, and 2) functional descriptions, performance characteristics, and development status for each component subsystem.

A "preliminary modular design" is the selection of the specific components that meet the composite requirements for all the missions considered and that may be interchangeably combined into given operational systems for specific applications. Such a design includes 1) block schematics of the complete complement of guidance and control components selected on the basis of the analysis leading to, and the evaluation of, the various conceptual designs, 2) functional desscriptions, physical characteristics, performance specifications, and interface characteristics for each of the modular elements, and 3) specification of the mechanical and electrical interfaces between the modular elements of the system and between the system and the launch vehicle or spacecraft.

The initial objective of Task III was to formulate the requirements for an integral modular guidance, navigation, and control system capable of meeting the mission requirements of Earth low-altitude polar and synchronous equatorial orbits, lunar orbit, Mars orbit, and solar probe (Jupiter flyby) missions. The results of Tasks I and II (see Ref. 1-1) provided the basis for this Task III formulation. Conceptual designs responsive to these requirements were then to be developed. Parametric variations of the performance characteristics of each of the critical components and subsystems of these conceptual designs were to be analyzed so as to permit the establishment of specific performance requirements relative to mission accuracy, fuel expenditure, system reliability, and weight. These analyses were to be used under the Task IV effort in specifying a "Preliminary Modular Design" and in assuring a technically sound rationale for the equipment specifications.

The study constraints and the scope of work can be summarized as follows:

a) The representative missions to be studied were

   1) Earth-Polar Orbit-Injection Mission utilizing Atlas/SLV3A/Burner II.

   2) Synchronous Equatorial Earth-Orbit Mission utilizing the Atlas SLV3X/Centaur. (Both direct ascent and parking orbit modes were to be considered.)

3) Mars Orbiter Mission (Voyager spacecraft launched by Saturn V).

4) Lunar Orbiter Mission (Lunar orbiter spacecraft launched by Atlas SLV3C/Centaur).

5) Solar-Probe Mission using Jupiter assist (advanced planetary probe spacecraft launched by Saturn IB/Centaur). (Close-in solar probe (0.1 AU) and out-of-ecliptic missions were to be considered.)

b) The resultant guidance and control instrumentation for a given set of launch vehicles, upper stages, unmanned spacecraft, and missions was to be based upon the boost phase (launch through injection) requirements as well as those for midcourse, target approach, encounter, and deboost into orbit phases of flight.

c) The choice of inertial systems was to be limited to strapdown systems.

d) Only the existing NASA and DOD radio tracking systems were to be considered. (See Ref. 1-1.)

e) Specific control system design concepts and interfaces with existing boost-vehicle control system elements were to be established for each of the launch vehicles. No attempt was to be made to optimize the total control system design.

f) Onboard computational requirements (memory capacity, word length, and execution time) were to be established utilizing the NASA-ERC United Aircraft computer concept described in Ref. 1-2. Sizing studies were to be based on guidance equations previously developed by TRW as well as the control equations developed in this study.

g) Computer interfaces were to be defined with respect to the NASA-ERC UAC concept defined in Ref. 1-2. Interface hardware (input/output) preliminary design was to be accomplished, but no specific design information for the computer was required.

h) The planet tracker used in the approach guidance system for the Mars mission was to be the NASA-ERC Kollsman sensor currently under advanced development. This is the only practical approach to the problem of planet tracking that has been demonstrated to date.

1.2 STUDY ASSUMPTIONS

1.2.1 Mission Definitions and Requirements

To meet the primary objectives of this study, i.e., evaluation of the feasibility of the integrated modular design concept for the guidance and control of launch vehicles and spacecraft for future NASA unmanned space missions, it is necessary first to establish specific mission and vehicle characteristics and mission performance requirements in order to proceed with the analysis. The specific assumptions made are detailed below. It is not expected that the study conclusions are sensitive to these particular assumptions.

It is assumed that the guidance requirements for the missions studied are representative of a major portion of the total requirements for NASA unmanned missions in the next decade. However, mission objectives are not precisely defined at the present time and definitive payload characteristics are not available. Also, launch vehicle selections for the missions have not been firmly made, and definitive design data are not available on vehicle upper stage concepts currently in the planning and development stages. For these reasons, it was necessary to postulate, somewhat arbitrarily, a set of specific mission performance requirements, launch vehicle selections, and vehicle and payload characteristics.

For the same reasons as given above, it is not possible to present complete and definitive mission performance (accuracy) requirements for the guidance and control system. Consequently, some of the accuracy requirements presented in this report are based on mission requirements determined from past studies. As more definitive trajectory data and mission objectives become available, these requirements can be updated.

The formulation of functional requirements and generic-candidate guidance system configurations is also dependent on mission analysis, although not to the extent that the formulation of accuracy requirements is. The functional requirements and candidate configurations can, therefore, be discussed in terms general enough to be applicable to any reasonable contemplated mission plans.

1.2.2 Postulated Vehicle/Payload Combinations

For purposes of this study, specific launch vehicle/payload combinations were postulated for the five missions. Table 1-I summarizes the mission-related data pertaining to the launch vehicles and the location of the radio/optical/strapdown inertial (ROI) guidance and control system for each of these missions. The table also delineates the specific TRW assumptions made relative to the guidance regime.

1.2.3 Upper Stage and Spacecraft Characteristics

Widely accepted quantitative values do not yet exist for upper stage (kick stage) or spacecraft weights, mass ratios, propulsion capabilities (thrust, specific impulse), and ΔV (velocity increment) capabilities for the missions under consideration. Without these, it is impossible to define with certainty the accuracy requirements for any mission phase or midcourse correction velocity limits. Lacking these data, it has been decided 1) to draw on results from other related studies (Refs. 1-3, 1-4, and 1-5) as much as possible or 2) to present the requirements in parametric form.

For the thrusting and ΔV capabilities, it has been assumed, for the lunar and interplanetary missions, that the spacecraft in which the radio/optical inertial guidance and control system is located has the necessary propulsion capability for accurately making corrective ΔV applications ranging from a few meters per second up to 100 m/sec. The highest thrust levels would be used for major orbital changes with ΔV values up to several thousand meters per second. It is also assumed that the spacecraft has complete three-axis control capability.

1.2.4 Separation of Guidance Functions Between Launch Vehicle/Kick Stage and the Mission Payload

For the earth orbiting missions it is assumed that the mission payload contains the capability for independent attitude control, propulsive maneuvers for small orbital corrections, communications, etc. At the time of separation of the payload spacecraft from the launch vehicle in which the ROI guidance system is located, these functions are activated and the ROI guidance and control functions terminated. Since the payloads

TABLE 1-I

RADIO/OPTICAL/STRAPDOWN INERTIAL TASK III MISSION SUMMARIES

Mission	Trajectory Characteristics	Booster	Guidance Package Location	Approximate Payload Weight	Guidance Regime
Earth Low-Altitude Polar Orbit	WTR Launch; ~927 km near polar circular orbit	Atlas SLV-3A/ Burner II	Burner II	2,500	Launch through insertion of payload into desired earth orbit (payload assumes orbit trim and stationkeeping functions after separation from last booster stage)
Earth-Synchronous Orbit a) Direct Ascent b) Parking Orbit Ascent	Same as used in Tasks I and II (See Ref. 2-1)	Atlas SLV-3C/ Centaur	Centaur	400	
Lunar Orbiter	Same as used in Tasks I and II	Atlas SLV-3X/ Centaur	Payload	2,000	Launch through insertion into final desired selenocentric or areocentric orbit, including all midcourse corrections and orbit change maneuvers
Mars Orbiter, 1975 a) Type I Trajectory b) Type II Trajectory	Saturn V injects spacecraft with desired C_3 into Type I or Type II interplanetary trajectory; spacecraft performs M/C and deboost into 1100 x 10,000-km orbit and subsequent injection into 500-km orbit	Saturn V	Payload	40,000	
Jupiter Flyby a) 0.1 — AU Probe b) Cross Ecliptic Probe	S-IB/Centaur injects spacecraft onto a high-energy interplanetary trajectory (C_3 = 121 km^2/sec^2, T = 464 days) post Jupiter trajectory determined by targeted $\bar{B} \cdot \bar{T}$, $\bar{B} \cdot \bar{R}$	Saturn IB/ Centaur	Payload	800	Launch through injection into interplanetary orbit and pre-encounter midcourse correction(s); post-encounter attitude control only

for three missions studied have not been defined in detail, reasonable assumptions have been made based on current spacecraft design trends.

1.3 STUDY IMPLEMENTATION

1.3.1 Tasks I and II Effort

The total study effort summarized in this volume was carried out in two distinct contract phases. The first phase, referred to as the Tasks I and II effort, was carried out in the six major steps listed below.

1) Functional and performance requirements for the strapdown inertial guidance subsystem and the electro-optical sensors were defined by mission phase for each of the four generic missions studied.

2) A survey was accomplished of state-of-the-art electro-optical sensors and strapdown inertial components (gyros and accelerometers) that potentially could be used.

3) Based upon the results of 1) and 2), appropriate candidate sensors were selected and performance (error) models were developed for them.

4) A study of possible radio guidance concepts and the capabilities of existing NASA and DOD tracking systems was conducted to define candidate systems, their applicability, limitations, and performance capabilities for the four missions.

5) An overall radio/optical/strapdown inertial guidance system concept, equipment configurations, and operating sequences were developed for each of the four mission categories.

6) Performance analysis studies were conducted both to investigate the performance capabilities of the candidate radio/optical/strapdown inertial guidance configurations and to demonstrate their adequacy for the four missions.

The detailed results of these studies is presented in Ref. 1-1 (vols. I through IV).

1.3.2 Tasks III and IV Effort

The second phase of the contract, referred to as the Tasks III and IV effort, may be divided into two groups of tasks:

- Derivation of guidance and control functional and performance requirements
  - Definition of mission characteristics
  - Conceptual design
  - System performance analyses
- Preliminary modular design
  - System configuration and interfaces
  - Subsystem design studies
  - Performance analyses of modular design

The Tasks III and IV studies extend and refine the results of the previous study effort. Vol. II of this report contains the detailed study results obtained under the Tasks III and IV effort. The following paragraphs describe briefly the implementation of each of these groups of tasks.

1.3.2.1 Derivation of Guidance and Control Functional and Performance Requirements (Task III)

1.3.2.1.1 Mission Characteristics (Vol. II Sec. 2)

Reference trajectories for the five basic missions were developed by TRW through the use of its Multivehicle N-Stage (MVNS) and Space Navigation Simulation (SNS) precision integration programs (Refs. 1-6 and 1-7). Trajectories generated under the previous study effort (see Ref. 1-1) were used for the earth-synchronous-orbit and lunar-orbit injections missions, utilizing the Atlas/Centaur launch vehicle. New trajectories generated under Task III were as follows:

Reference Powered Trajectories

a) Atlas Burner II — Low altitude earth-circular polar-orbit mission launched from WTR.

b) Saturn V — Launch-to-injection trajectory, with earth-injection conditions determined to match the interplanetary trajectories defined below.

c) Saturn IB/Centaur — Launch-to-injection trajectory with earth-injection conditions chosen to match the interplanetary trajectories defined below.

Reference Interplanetary Trajectories

a) Mars Orbiter Missions — Based on trajectory and mission analyses conducted for the Mars 1975 launch opportunity under the TRW Voyager Task D study (Ref. 1-5), Types I and II reference trajectories were selected for the two Mars orbiter missions to be considered. The rationale for selection of the reference trajectories is presented together with a comparison of the heliocentric trajectory characteristics of both mission types in sec. 2 of this volume.

Injection state vectors for these Mars missions were computed analytically assuming a Saturn V launch vehicle and a 100 n. mi., short coast, circular parking orbit.

b) Jupiter Flyby Missions — Trajectory data for Jupiter flyby missions during the 1972 launch opportunity were generated for the two specified flyby missions. Reference trajectories were selected and an analytic computation of the injection state vector was performed assuming a Saturn IB/Centaur launch vehicle and a 100 n.mi., short coast, circular parking orbit.

In addition to generation of the analytic state vector required at injection, the vehicle's position with respect to the sun, earth, target planet, and Canopus was determined for all reference trajectories analyzed. Time histories of these quantities were developed for both the near-earth and heliocentric phases of the missions. Target planet approach geometry was defined for all reference trajectories and capture conditions and orbit orientation geometry were developed for the Mars orbit missions.

1.3.2.1.2 Guidance Control Conceptual Designs Vol. II Sec. 3)

The implications of mission objectives on variable versus fixed time-of-arrival midcourse guidance schemes were examined for the Mars Type I mission, including tradeoffs between midcourse correction capabilities and requirements, and for the particular guidance schemes.

Earth-based tracking and computation was established as the primary navigation mode for the lunar and interplanetary missions and for the determination of orbital parameters for the Mars orbiter missions.

The booster and spacecraft attitude control system concepts were examined, and a digital system was selected as the most appropriate for the applications considered. Control system interface tradeoff studies were conducted to define the functional interfaces between the ROI guidance and control system and the existing or modified boost vehicle control electronics and thrust vector and reaction control systems.

Special emphasis was placed on studies relating to attitude-fixed versus attitude-maneuvering spacecraft/payloads and gimbaled versus fixed optical sensors. For the translunar and interplanetary coast phases, body-fixed optical sensors were selected as the most appropriate spacecraft attitude references. For the Mars approach guidance, high-precision optical measurements are required; gimbaled Canopus and planet sensors were chosen as the most appropriate for this application.

For each mission/booster/payload, an overall functional description and schematic of the radio/optical/strapdown inertial guidance system were developed. These include the general signal flow, and moding and switching functions. Detailed mechanization equations were defined as required to define the data flow between subsystems and the operational moding and sequencing functions.

1.3.2.1.3 Guidance and Navigation Performance Analyses (Vol. II Sec. 4)

a) Sun-Sighting Time-Updating Technique — The time-updating technique for the multiparking orbit earth-synchronous missions was analyzed in detail. The accuracy of this method and the impact on overall system accuracy were assessed.

b) Powered Flight Performance Analysis — For those missions in which the guidance, navigation, and control system under study has prime control over the boost and injection phase, the GEAP II error analysis program (Ref. 1-8) was used to evaluate injection accuracy and to establish the requirements for midcourse velocity corrections. Parametric tradeoff studies involving strapdown inertial instrument quality, and prelaunch azimuth alignment errors, were performed. Midcourse correction, deboost maneuver, and orbital transfer maneuver accuracies were also evaluated.

c) Interplanetary and Approach Navigation Analysis — The SVEAD program (Ref. 1-9) for estimating navigation accuracy was modified to give it the capability of handling closed orbits around Mars. The analyses made earlier under Task II for the Mars mission were extended both to incorporate variations in optical sensor accuracies and to examine the implications of Type I versus Type II trajectories.

1.3.2.1.4 Control System Performance Analyses (Vol. II Sec. 5)

Bending modes were generated for the Saturn V/Voyager vehicle configuration, and existing bending data, propellant sloshing data, aerodynamic and mass properties data, and thrust vector control characteristics for each launch vehicle were assembled for use in subsequent control system analyses (see Vol. II, apps. C, D, and E).

Stability margins of the linear control system for the first stages of the selected boost-vehicle configurations were determined (see app. A of vol. 2). A comparison was made between the use of first-stage rate gyros and upper-stage gyros, and the digital compensation required under these conditions established. Stability margins for the Voyager spacecraft were also determined.

Coast-flight attitude-reference acquisition, maneuvers, and normal mode operations were analyzed.

1.3.2.2 Preliminary Modular Design (Task IV)

Preliminary modular designs were developed for each mission based on the conceptual designs. Interface definitions were established for the onboard computer; the control system; and the telemetry, tracking, and command system. Detailed equipment descriptions and specifications were developed for the electro-optical sensors and the inertial reference unit.

1.3.2.2.1 System Configuration and Interfaces (Vol. II Sec. 6)

a) Modularity Concept — An equipment modularity concept for the total radio/optical/strapdown inertial guidance system was established in accordance with the basic ROI Study objectives. For each of the missions, TRW established the equipment utilization concept, and defined the interconnections and interfaces of the various units comprising the system.

b) Vehicle Interfaces and Mechanical Mounting Considerations — Physical locations and interconnections of the modular radio/optical/strapdown inertial guidance system components were established for each of the five launch vehicle/mission combinations, considering optical sensor line-of-sight requirements and other location and mounting constraints. Interfaces with existing vehicle control system elements were established in accordance with the control system conceptual and modular design studies.

c) Guidance Equipment Mechanical Interface and Packaging Considerations — Sensor mounting provisions (necessary for adequate mounting stability) were established including the requirements for precision navigation base assemblies and an electronics packaging modular design concept.

d) Thermal Design Considerations — For each mission, the expected thermal environment conditions and constraints were established for the guidance and control equipment at the appropriate location in the launch vehicle, upper stage, or spacecraft. A survey was conducted to establish the approximate operating temperature range for the most critical optical sensors, and thermal control concepts were established to the extent possible using available design data on various boost vehicles and spacecraft.

   Temperature control requirements and concepts were established for such units as the IRU, where the required performance can be achieved only through precise thermal control of critical elements.

1.3.2.2.2 Onboard Computational Elements (Vol. II Sec. 7)

Onboard computational requirements (memory size, word length, and speed requirements) were established for the NASA-ERC/UAC computer concept (Ref. 1-2). These studies were based on equations previously developed by TRW for the LM Abort Guidance System (attitude reference and navigation computations) (Ref 1-10), Advanced Centaur Studies (steering and guidance computations) (Ref 1-11) plus the control equations developed in this study.

Major emphasis was placed on defining in detail the I/O interfaces between the computer and the electro-optical sensors, the inertial reference unit, the control system components, and the telemetry, tracking, and command subsystems. A conceptual design of a computer interface unit (CIU) was developed providing interface compatibility with the NASA-ERC/UAC Advanced Kick Stage Guidance Computer (Ref. 1-2). A reliability estimate for this computer was developed for use in mission reliability studies.

1.3.2.2.3 Control System Design (Vol. II Sec. 8)

Tradeoffs were made between control system digital autopilot equation complexity and computational-time and memory-storage requirements. Several digital compensation filters were considered to determine the cost of added flexibility in the digital control system.

The interface between the computer and the control system hardware was defined with considerations given to signal levels issued to the thrust-vector actuation system and to the receipt of signals from interfacing gyro packages. An evaluation was made of the signal mixing requirement, either within or outside the computer for differential roll control; the problem of interfacing with a varying number of boost-vehicle engines was also addressed.

Functional schematics of the Atlas/Centaur, Saturn V/Voyager, and Saturn IB/Centaur digital control systems were generated showing signal flow, and moding and switching functions.

1.3.2.2.4 Electro-Optical Sensor Designs (Vol. II Sec. 9)

The optical sensors selected under Tasks I and II (Ref. 1-1) were reviewed both in light of recent state-of-the-art developments and of new requirements resulting from present mission specifications. Specifically, the applicability of gimbaled Canopus and planet approach sensors and the use of a very narrow field sun sensor were considered.

Based on this review, TRW chose a set of sensors appropriate to the study effort and established a configuration for each mission. Sensor specifications were prepared covering functional description, accuracy, physical performance, and reliability. A description of data interface characteristics and the accuracy of the sensor configurations were generated to support the guidance accuracy studies. The state of development of each of the sensor elements was evaluated.

For each sensor required by the several missions, a preliminary design was generated using available data on existing equipment, where applicable, plus additional preliminary design effort as required. The following characteristics were established for each sensor:

a) Sensor operating modes

b) Sensor accuracy

c) Final data interface characteristics

d) Weight, dimensions, electrical power requirements

e) Sensor reliability models and numerical parameter

f) Mechanical and electrical mounting characteristics consistent with required physical interchangeability

g) Physical description consisting of a preliminary design drawing for each sensor.

1.3.2.2.5 Inertial Reference Unit (Vol. II Sec. 10)

A preliminary design was generated of a strapdown IRU meeting the performance requirements of the several missions based on the previous studies carried out under Tasks I and II (Ref. 1-1). The following characteristics were established for this unit:

a) Sensor and electronics accuracy including environmental sensitivities (linear and rotational acceleration and vibration environments)

b) Data interface characteristics

c) Weight, dimensions, electrical power requirements

d) Mechanical mounting characteristics

e) Mechanical electrical packaging and thermal control concepts

f) Reliability estimate.

1.3.2.2.6 Performance Characteristics of Modular Design (Vol. II Sec. 11)

A performance analysis summary for the preliminary modular design was established based on the recommended sensor selections and specifications demonstrating that the preliminary modular design satisfies the guidance and control requirements for the five missions studied. The overall system performance characteristics were related to trajectory accuracy and fuel required for correction of guidance, navigation, and control errors.

Weight, power, and total failure-rate estimates were made for each of the elements comprising the modular system and the results used to estimate the overall system reliability, weight, and power for each of the five missions considered.

1.4 DEFINITION OF TERMS

Certain of the definitions pertaining to the missions, the launch vehicle, mission events, and trajectories used throughout this report are summarized below.

1.4.1 Missions

In general, the term "mission" is used in this report to encompass and describe the events which are associated with directing the launch vehicle or the spacecraft from the earth and which terminate with the accomplishment of the mission objectives. In the analysis of the various missions described in the ROI Study, the following terms are used:

Synchronous Earth Orbit Mission	In the synchronous earth orbit mission, the launch vehicle is used to place the satellite payload into an earth-synchrouous (24-hr period) equatorial orbit at a desired longitude. The injected payload (satellite) is assumed to have orbit trim and station-keeping capability.
Orbiter Missions	In an orbiter mission, approximately at the time when the spacecraft is closest to the target body (moon or planet), its trajectory is deliberately altered by a propulsive maneuver so that it remains in an orbit about the target body as a satellite.

Solar Probe Mission	In a solar probe mission the spacecraft is injected into a heliocentric orbit that passes within a specified distance of the sun. This is an untargeted mission requiring no trajectory alterations subsequent to injection.
Flyby Mission	In a flyby mission, the spacecraft passes close to the target planet. No propulsion forces are employed to alter the trajectory so as to remain in the vicinity of the target planet. The spacecraft departs from the region of the target planet, although its trajectory will have been perturbed.
Solar Probe with Planetary Swingby	In this type of mission the spacecraft passes close to a planet with the purpose of significantly altering the spacecraft trajectory. After departure from the target planet, the spacecraft continues on a heliocentric trajectory to within a prescribed distance from the sun. No propulsive forces are employed to alter the trajectory in the vicinity of the target planet. For a given distance of closest approach to the sun, this technique may be used to significantly reduce the launch vehicle ΔV requirements, usually at the expense of considerably longer mission durations.

1.4.2 Vehicle Terms

Launch Vehicle	The launch vehicle includes the multistage boost vehicle which injects the spacecraft into the desired trajectory and includes all hardware up to the interface where the spacecraft is mated and the payload shroud attaches which protects the spacecraft. Generically, the launch vehicle system also includes all appropriate ground support and test equipment.

Kick Stage	For the purposes of this study, "kick stage" refers to the final powered stage of the launch vehicle (the payload spacecraft is assumed to have only limited velocity capability for incremental orbit corrections). The kick stage is assumed to provide complete three-axis guidance, navigation and control capability for all launch vehicle stages except for the Saturn V (Mars orbiter mission).
High Energy Upper Stage (HEUS)	This is a particular kick stage concept using an advanced propulsion system burning high-energy propellants such as H_2/F_2. Typical gross weight is 3200 kg. The thrust to weight ratio is approximately 1.
Spacecraft	The spacecraft system encompasses the payload itself and all its component subsystems, the science payload, the adapter which is mounted to the kick stage, and limited propulsion capability for orbital corrections.
Launch Operations System	The launch operations system does not include any flight hardware, but constitutes the operational responsibility for supporting and conducting the launch of the combined launch vehicle and spacecraft through the separation of the spacecraft from the launch vehicle.
Mission Operations Systems	Operational responsibility for supporting and conducting the mission after the spacecraft is separated from the launch vehicle is borne by the mission operations systems.

1.4.3 Mission Events

In the analysis of the various mission events described in the ROI Study, the following terms are used:

Prelaunch	Collectively, all events before liftoff.
Launch	Collectively, all events from liftoff to injection.

Liftoff and Ascent	Departure of the combined launch vehicle-spacecraft from the ground and ascent to a parking orbit of specified altitude (typically 185 km (100 n. mi).
Injection (synchronous earth orbit mission)	Thrust termination of the kick stage, placing the kick stage/payload into a transfer trajectory to synchronous altitude from the parking orbit or, alternately, into the final synchronous earth orbit.
Injection (lunar or interplanetary mission)	Thrust termination of the lower stages of the launch vehicle, placing the kick stage/payload into an interplanetary or translunar trajectory, from the parking orbit.
Separation (shroud)	Detachment of the nose fairing from the launch vehicle during ascent.
Separation (spacecraft)	Detachment of the spacecraft from the spacecraft kick stage adapter after injection.
Orientation Maneuver	A programmed alteration of the injection stage or spacecraft attitude to cause it to return to a desired orientation, such as the cruise orientation.
Midcourse Trajectory Correction Maneuver	A propulsive maneuver performed to compensate for inaccuracies or perturbations so as to redirect the spacecraft toward the intended aiming point. Generally, it requires orientation to a specific attitude, operation of the rocket engine, and reorientation to the cruise attitude. The time of this maneuver is during the interplanetary or translunar flight, but not necessarily at the midpoint.
Encounter	Generally, encounter encompasses events occurring when the spacecraft is near the target planet. Specifically, it refers to the time when the spacecraft is at its point of closest approach (periapsis).

Orbit Insertion	The propulsive braking maneuver by which the (orbiter) spacecraft trajectory at the target planet is changed from approach (hyperbolic) to orbital (elliptical).

1.4.4 Trajectory Terms

In discussing the trajectories possible for the various missions of the ROI Study, the following terms are used:

Direct Trajectory	An interplanetary trajectory from the earth to a target planet, in which no intermediate planets (or satellites) are approached closely enough to significantly influence the trajectory.
Swingby Trajectory	An interplanetary trajectory from the earth to a target planet, in which an intermediate planet is passed sufficiently closely to exploit the effect of its gravitational attraction. This exploitation may provide reduced mission duration, reduced launch energy, or an opportunity for scientific observations of the intermediate planet.
Launch Opportunity	The time during which trajectories to a target planet may be initiated from the earth, with reasonable launch energies. A launch opportunity is usually identified by the year in which it occurs, and the target planet.
Launch Period	The space in arrival date-launch date coordinates in which earth-planet trajectories are possible in a given launch opportunity; specifically, the number of days from the earliest possible launch date to the latest.
Launch Window	The time in hours during which a launch is possible on a particular day.
Geocentric (heliocentric, planetocentric)	Described or measured with respect to inertial coordinates centered with the earth (sun, planet). Pertaining to the portion of the flight in which the trajectory is dominated by the gravitation of the earth (sun, planet).

C_3, Launch Energy, Injection Energy	Twice the geocentric energy-per-unit mass, of the injected spacecraft. This is equivalent to the square of the geocentric asymptotic departure velocity.
Asymptote	The line that is the limiting position which the tangent to a hyperbolic (escape) trajectory approaches at large distances from the attracting center.
DLA	Declination of the outgoing geocentric launch asymptote.
ZAL	Angle between the outgoing geocentric asymptote and the sun-earth vector.
ZAP	Angle between the incoming planetocentric asymptote (at the target planet) and the planet-sun vector.
ZAE	Angle between the incoming planetocentric asymptote (at the target planet) and the planet-earth vector.
V_∞ or V_{HP}	Planetocentric asymptotic approach velocity.
Parking Orbit	An unpowered, geocentric, approximately circular orbit, separating the powered portions of the launch and injection sequence.
Type I, Type II Interplanetary Trajectories	Type I transfers are defined as those in which the vehicle traces a central angle of less than 180° about the Sun between departure from the Earth and arrival at Mars. In Type II trajectories, the angle is greater than 180°.

1.4.5 Coordinate Systems

The various coordinate systems used in specifying performance requirements and powered flight performance analysis results obtained during the ROI Study are defined as follows:

ECI (Earth-Centered-Inertial)	This is a right-handed coordinate system, in which Z lies along the earth's polar axis and X and Y lie in the earth's equatorial plane. The

X-axis passes through the Greenwich meridian or in the direction of the Vernal Equinox at the time of launch, (specified in text).

RTN (Radial-Tangential-Normal) — A right-handed orthogonal coordinate system in which R lies in the direction of the nominal position vector from the center of the earth, and N lies in the direction of the orbital angular momentum. T forms a right-handed orthogonal set with R and N.

(X, Y, Z) Selenographic — Moon-Centered Inertial Coordinates. This is a right-handed orthogonal coordinate system in which Z lies along lunar polar axis, and X, Y lie in the lunar equatorial plane with X passing through zero lunar longitude (Sinus Medii).

$\overline{B} \cdot \overline{R}$, $\overline{B} \cdot \overline{T}$ — For a given interplanetary trajectory, the impact parameter vector $\overline{B}$ specifies in which direction from the planet and what distance the approach asymptote lies. $\overline{B}$ is commonly expressed in components $\overline{B} \cdot \overline{R}$ and $\overline{B} \cdot \overline{T}$, where $\overline{R}$, $\overline{S}$, $\overline{T}$ are a right-hand set of mutually orthogonal unit vectors aligned as follows: S is parallel to the planet centered approach asymptote, $\overline{T}$ is parallel to the plane of the ecliptic and positive eastward, and $\overline{R}$ completes the set and has a positive southerly component.

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2. SUMMARY OF MAJOR STUDY CONCLUSIONS AND RECOMMENDATIONS

The principal conclusions and recommendations resulting from the study are summarized in this section. Subsec. 2.1 presents our conclusions relative to the overall guidance and control system concept and configuration for each of the missions. Subsec. 2.2 presents our conclusions relative to the modular design of the onboard optically aided strapdown inertial guidance and control system. Subsec. 2.3 summarizes the principal conclusions of the portion of the study dealing with the application, limitations, constraints, and performance capabilities of radio guidance.

2.1 GUIDANCE AND CONTROL SYSTEM CONCEPT

2.1.1 General Conclusions

In this study, the applicability of state-of-the-art guidance concepts utilizing appropriate combinations of ground-based radio tracking and onboard inertial and optical sensors has been evaluated for five representative missions (see sec. 3 of this volume for a description of the mission characteristics). The following general conclusions relative to the system concept were reached:

1) The guidance functions for the five missions can feasibly be accomplished in an efficient manner by utilizing appropriate combinations of navigation sensors consisting of ground-based radio tracking and onboard inertial and optical sensors. The concept of the radio/optical/inertial guidance system evolved during this study consists of a "core" strapdown inertial subsystem (inertial reference unit and computer) with the capability of adding appropriate electro-optical sensors (star trackers, horizon sensors, sun sensors, etc.) to tailor the system for a particular mission application. The onboard system also includes a transponder and data link working in conjunction with the ground-based tracking systems.

2) The control functions for the five missions, for the launch vehicles and spacecraft studied, can be accomplished efficiently by utilizing the onboard inertial reference and digital computer together with suitable control electronics packages for interface compatibility with the launch vehicle and spacecraft thrust vector

control and reaction control systems. In this concept the onboard inertial system is used as a short-term attitude reference, and suitably chosen electro-optical sensors are used for long-term attitude references.

3) The concept of a modular guidance and control system that meets the composite requirements of the missions studied is feasible, and is, therefore, an attractive means to implement the guidance and control requirements. This has been demonstrated by the successful development of a preliminary modular design that meets the performance requirements for each of the missions and the interface requirements for each of the launch vehicles and spacecraft.

4) For the synchronous earth-orbit mission, the guidance functions (launch-through-final-orbit insertion) can be performed efficiently by the onboard inertial system, supplemented by optical aids for attitude and position updating during long coast periods. Radio tracking may also be used as an alternate method of position updating, however, severe operational and mission constraints are encountered, which make its use as the primary guidance system unattractive.

5) For the interplanetary missions, radio guidance (i.e., ground-based radio tracking and orbit determination) is essential to meet the mission objectives and is the only reasonable method* of meeting the demanding mission performance requirements. The concept recommended here uses the existing NASA Deep Space Instrumentation Facility (DSIF) as the primary means of orbit determination during the interplanetary trajectory phases. Powered maneuvers for trajectory correction and insertion into orbit around the target body are performed under control of the onboard optical/inertial system.

6) For the lunar mission, it is concluded that the most reasonable approach is to use the NASA Unified S-Band (USBS) tracking system as the primary navigation sensor, and to use the onboard optical/inertial system for controlling the powered maneuvers in a manner

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*Although it is theroretically possible to perform the interplanetary missions using a completely autonomous, onboard, optically aided, inertial system (no ground-based radio tracking), this approach places severe accuracy requirements on the onboard system (particularly the optical sensors) and requires significantly greater fuel allowances for performing trajectory-correction maneuvers.

similar to that for the interplanetary missions. Although the performance requirements for an autonomous system would be considerably less severe than those for interplanetary missions, it would be very difficult to achieve the navigation accuracies attainable with the presently existing radio-tracking systems.

7) Radio guidance is of limited utility for boost-phase (launch-through-orbit insertion) guidance for the launch vehicles and trajectories considered. The major problem is the limited tracking system coverage from the existing tracking stations particularly for missions requiring a parking-orbit coast phase.

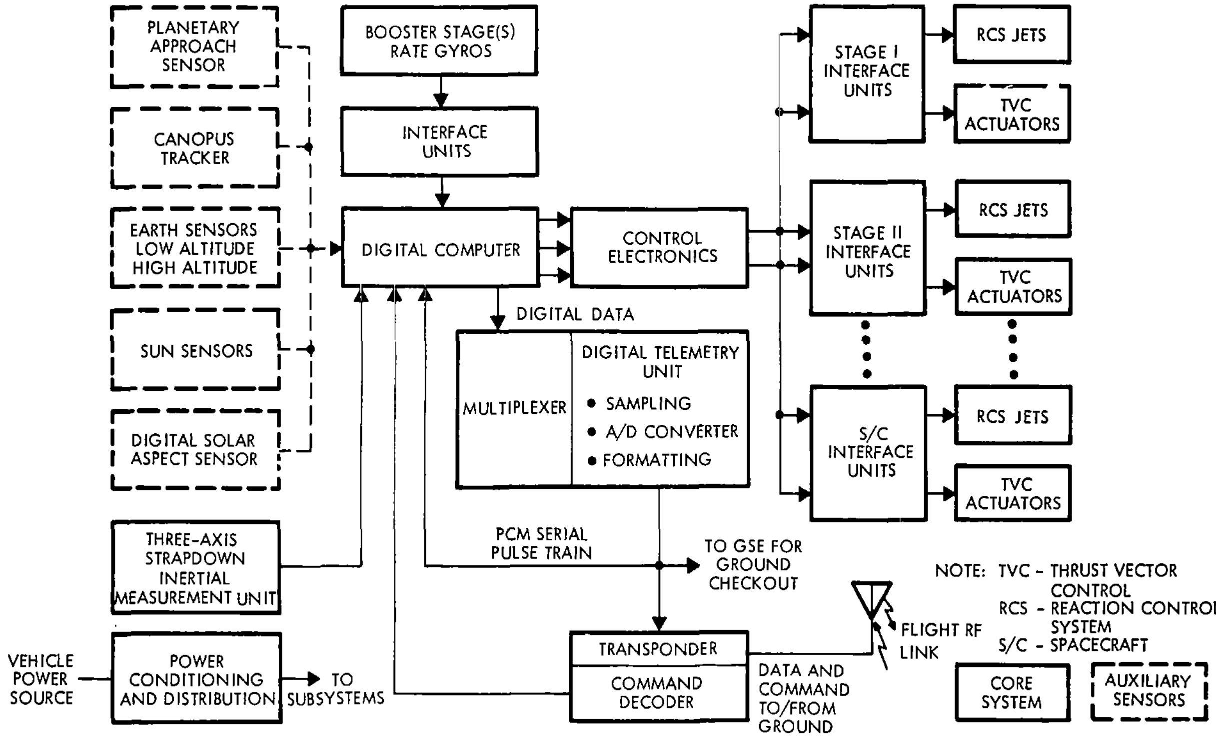
8) Boost-phase guidance (launch-through-orbit insertion including the parking-orbit coast phase) may be performed with sufficient accuracy, using only the onboard inertial system. However, onboard optical sensors are required in the extended coast phases (earth-orbit, translunar, or interplanetary coast) for correcting the attitude drift rate of the onboard inertial sensors (gyros), or as the primary attitude reference.

9) If the onboard system is located in the final stage of the vehicle or in the payload spacecraft, it is feasible from a functional and performance point-of-view to use it for guidance of the lower stages starting at liftoff, provided that the interface between the guidance system and the vehicle control system is properly configured. The evaluation of the desirability of using a single guidance system or, alternately, a separate system for lower-stage guidance and control, depends on vehicle and program considerations not considered in this study.

10) The strapdown IRU together with the digital computer and the vehicle control system provides a precision capability for performing powered maneuvers for the midcourse trajectory correction, orbit insertion, and orbit trim maneuvers required by the missions. This system is capable of providing preburn attitude maneuvers, closed-loop steering during the propulsive phase, and accurate thrust cutoff based on the velocity (ΔV) accumulated during the burn.

2.1.2 Recommended Equipment Configuration by Mission

A block diagram of the total guidance and control system suitable for any of the missions is shown in Figure 2-1, together with a matrix, Table 2-I, showing the specific equipment utilization by mission. The recommended configuration is that of a basic "core" system used for all the missions, with auxiliary sensors added in a modular or building-block fashion



COMMAND LINK USED FOR FLIGHT AND GROUND SEQUENCING, MODE CONTROL, ETC.
TELEMETRY LINK (HARD WIRE AND/OR RF) USED FOR GROUND CHECKOUT

Figure 2-1. Composite Equipment Configuration

TABLE 2-I

EQUIPMENT UTILIZATION

Equipment		Near-Earth Polar Orbit	Earth-Synchronous Orbit	Lunar Orbit	Solar Probe (Jupiter Swing-by)	Mars Orbiter
Core System	3-axis strapdown inertial measurement unit	▲	▲	▲	▲	▲
	Digital Computer	▲	▲	▲	▲	▲
	S-band tracking transponder and command link		▲	▲	▲	▲
	Auxiliary equipment power conditioning and distribution, telemetry, etc.	▲	▲	▲	▲	▲
Auxiliary Sensors	Star (Canopus) tracker (attitude reference)			▲	▲	▲
	Earth sensor (horizon scanner) (local vertical reference) • Low Altitude • High Altitude		▲			
	Sun sensor (cruise attitude) reference)			▲	▲	▲
	Sun sensor solar aspect sensor for attitude reference and navigation fix (optional)		▲			
	Planetary approach sensor					▲

to configure the system to a particular mission. The auxiliary sensors interface with the core system through the digital computer. If the computer input/output design is such as to accomodate any set of auxiliary sensors without any required redesign, then the mission-dependent changes can be accomplished with a minimum of effort by suitably changing the stored computer programs (software).

While the implementation of the "core" inertial guidance and control system is identical in each of the mission, its role varies significantly from mission to mission. For example, in the synchronous earth orbit mission, the strapdown subsystem (supplemented by appropriate electro-optical sensors) can essentially provide complete autonomous guidance and navigation. In the lunar orbit mission, it provides precise guidance during boost and translunar orbit injection, and midcourse and orbit-insertion maneuvers with primary translunar navigation provided by ground tracking during the coasting phases. The inertial subsystem provides primary attitude reference information for the synchronous earth orbit mission; in the other missions, primary attitude-reference information during heliocentric orbit phases is provided by the sun and star sensors.

The inertial measurement unit is a strapdown configuration. Outputs of the three orthogonal body-mounted gyros are in the form of pulses, each quantized pulse representing an incremental attitude change about the gyro's sensitive axis. The computer accepts this information and can generate body angular-rate information and/or total body-attitude information. The output pulses of the three body-mounted accelerometers represent velocity increment information, which is combined with the gyro data to provide total velocity change information in some chosen set of inertial reference axes. A detailed description of the strapdown inertial subsystem is presented in sec. 10, vol. II, and is summarized in subsec. 7.1 of this volume.

The auxiliary sensors in this study were limited to electro-optical sensors, and used primarily for attitude referencing and planetary approach navigation. These sensors include earth horizon scanners, sun sensors, star trackers, and planet sensors. The application of these sensors for each mission is discussed in detail in sec. 3. Descriptions and performance characteristics of individual sensors are presented in vol. II, sec. 9, and are summarized in subsec. 7.2 of this volume.

2.1.3 Utilization of the Strapdown Inertial Reference Unit

Performance studies were carried out using TRW's GEAP II Generalized Inertial Guidance Error Analysis Digital Computer Program (Ref. 2-13), using the error models developed for two strapdown IRUs. The results of these performance analyses are summarized in sec. 5 and are presented in detail in sec. 7, Ref. 2-1 and sec. 4, vol. II. On the basis of these performance studies, it is concluded that:

1) The five specified missions can be accomplished utilizing either of the IRUs postulated. Use of optical sensors is required to correct for the attitude drift rate of the strapdown IRU over extended coast periods in all of the missions.

2) Boost-phase guidance (launch through initial parking-orbit insertion) may be performed satisfactorily by a strapdown system.

3) The strapdown IRU may be used as a short-term vehicle-attitude reference during coast phases. For short (less than one orbit) parking-orbit coasts, no auxiliary sensors are required. For longer parking-orbit coast times and for translunar and interplanetary cruise phases, the inertial attitude reference provided by an auxiliary set of optical sensors is required.

2.1.4 Utilization of Electro-Optical Sensors

As part of the conceptual design, mission analyses and performance studies have been carried out to determine the functional and performance requirements for electro-optical sensors. The results of these analyses are summarized in secs. 4 and 5 of this volume and are presented in detail in secs. 3, 4, and 9 of vol. II. On the basis of these mission analyses and performance studies, it is concluded that:

1) The five specified missions can be accomplished by utilizing various combinations of sun sensors, earth sensors, Canopus sensors, and a planetary approach sensor.

2) For attitude updating during long-coast phases in earth orbit, an earth sensor (horizon scanner) and sun sensor combination is recommended.

3) A time updating scheme using a sun sensor is recommended for the earth-synchronous satellite mission (See vol. II, subsec. 4.3.) This scheme offers a simple solution requiring a minimum of onboard computational complexity, and obviates the need for ground-based radio tracking for position updating prior to the final injection maneuver.

4) For the translunar and interplanetary cruise phases, a long-term inertial attitude reference is required. For the missions considered in this study, a sun sensor/star (Canopus) sensor combination is recommended.

5) With the exception of the planet approach phase of the Mars mission, the electro-optical sensor performance requirements can be met by utilizing presently available instrument designs.

6) For the Mars approach phase, the highest accuracy attainable is desired in measuring the sun-planet and Canopus-planet angles. Use of a gimbaled planet tracker and Canopus tracker is desirable to achieve the desired performance.

7) Currently available gimbaled star trackers representing the most recent advances in the state-of-the-art will meet the functional and performance requirements for the Mars approach phase, (however, a precision gimbaled planet tracker must be developed. The planet tracker, currently in the early stages of development by NASA-ERC Kollsman and employing a rim scanning technique, can probably be developed to meet the functional and performance requirements for this sensor.

2.1.5 Control System Concept

Control system analytical design and interface studies have been carried out to define an integrated guidance and control system concept for the five missions under consideration. Both launch vehicle and upper stage/spacecraft control functions have been considered. The results of these studies are presented in secs. 3, 5, and 8 of vol. II. On the basis of these analyses it is concluded that:

1) A digital control system concept utilizing the guidance computer represents an efficient and flexible means of implementing the launch vehicle and spacecraft control functions. With proper gain and filter coefficient changes within the computer, the conventional control laws that are applicable to booster flight control can also be employed for spacecraft coast attitude control and spacecraft powered flight control.

2) The IRU provides a suitable attitude reference for control purposes for all stages of powered flight for all of the launch vehicles considered. For upper stages or spacecraft, the vehicle attitude data supplied by the IRU may also be used to derive vehicle angular rates by appropriate digital computations. During boost-phase flight, auxiliary angular rate sensing instruments are required on the boost vehicle to stabilize structural bending modes.

2.2 PRELIMINARY MODULAR DESIGN

Preliminary design studies of a modular radio/optical/strapdown inertial guidance and control system have been conducted for the five missions and associated launch vehicles defined in Table 1-I. The results of the modular design studies are summarized in sec. 6 of this volume and presented in detail in vol. II, secs. 6 through 11. On the basis of these studies, the following conclusions are drawn:

2.2.1 System and Subsystem Interfaces

1) A modular equipment concept meeting the functional and performance requirements of the five missions is a feasible concept, and represents an efficient means of implementing the guidance and control requirements for a given mission without hardware redesign. The equipment configuration for each mission and the interfaces with the launch vehicle are summarized in sec. 6.

2) The recommended modular guidance and control system consists of an assemblage of separately packaged elements electrically connected by an interconnecting harness. Major separable elements are the digital computer, inertial reference unit, electro-optical sensors, sensor electronics, control system electronics, and the telemetry, tracking, and command subsystem.

3) Typical vehicle mounting locations for the various elements of the modular system are indicated in subsec. 6.2. Mounting provisions (mechanical interfaces) for each item of equipment may be standardized. However, vehicle dependent integration hardware (mounting brackets, cable harnesses, etc.) must be provided for each installation.

4) To simplify the subsystem interfaces, it is desirable to provide only +28 Vdc unregulated power directly to each subsystem from a common power source. Secondary power supplies are provided, as required, within each subsystem.

5) With the exceptions noted below, the elements of the modular system may be mounted directly to the launch vehicle or spacecraft structure as indicated in subsec. 6.2. Optical access and thermal control requirements must be considered in the installation design. Passive thermal control techniques appear to be adequate for the mounting locations considered.

6) For the Mars mission, a precision mounting is required to provide the required alignment tolerances and stability among the approach guidance sensors and the inertial reference unit. A similar, but less accurate mounting base is required for the synchronous earth-orbit mission.

2.2.2 Onboard Computational Elements

Definition of the onboard computational elements of the preliminary modular design has been restricted in this study to two areas: 1) definition of the computational requirements for the modular system and 2) mechanization studies of the interface between the digital computer and other elements of the modular system. In accordance with the contract work statement, the preliminary design of the digital computer is specifically excluded from this study. Results of the studies made in these two areas are summarized in subsec. 6.4 and presented in detail in sec. 7, vol. II. Conclusions and recommendations with regard to this study area are as follows:

1) A digital interface unit is recommended to provide a suitable interface between the central digital computer and the other elements of the modular guidance and control system. The detailed interface requirements and the preliminary design for the interface unit are presented in sec. 7, vol. II.

2) The digital interface unit is designed to provide interconnections for all elements of the modular guidance system for all five mission applications. As such, it provides the flexible interconnection capability required of the system to meet the requirements of specific missions.

3) The total computational requirements for the modular guidance and control system, as implemented in the central computer, are quite reasonable and easily implemented by state-of-the-art aerospace computers. Although many computational functions are common to all missions, some mission-to-mission software changes will be required.

2.2.3 Control System Modular Design

The control system conceptual designs for each of the missions described in sec. 3 define the functional and interface requirements with existing launch vehicle control system components. Based on these conceptual designs, preliminary design studies have been conducted to define the characteristics of the interface hardware that may be regarded as part of the modular guidance and control system. In addition, a modular set of control equations has been developed applicable to all of the missions studied. The results of these studies are summarized in subsec. 6.5. Some general conclusions and recommendations based on these studies are as follows:

1) A variety of interface mechanizations is possible. These are characterized by the degree to which existing launch vehicle control system elements (sensors and electronics) are utilized. The recommended interface design is an intermediate approach in which the strapdown system provides an attitude reference for all stages of flight. Existing rate gyro packages are retained for the lower stages. Downstage control electronics packages are retained, but modified to eliminate unnecessary functions.

2) A control electronics package is required to provide appropriate signal interfaces between the guidance and control computer and a) the existing booster thrust vector control (TVC) systems and b) the upper stage/spacecraft TVC system and reaction control system (RCS). This package provides discrete signal power amplification, signal conditioning of thrust vector control commands, and signal switching and distribution functions.

3) Rate gyro packages are needed to stabilize the Saturn V and Atlas/Centaur vehicles. Although the rate gyro packages are desirable, they are not required to stabilize the Saturn IB vehicle. Because of the nine-tank cluster on the Saturn IB first stage, considerable structural stiffness exists and interactions between the control system and structural modes are correspondingly reduced.

4) For the Saturn V/Voyager configuration, a desirable location for the rate gyro package would be on the aft end of the Saturn S-II stage; however, the present location on the aft end of the S-IVB stage is considered acceptable. For the Atlas/Centaur configuration, relocation of the rate gyro package along the Atlas Booster for specific payloads would be desirable.

2.2.4 Sensors

Preliminary design studies have been conducted for the electro-optical sensors and the inertial reference unit defined by the guidance and control system conceptual designs for each of the missions (see sec. 4). Sensor descriptions, operating characteristics and performance specifications and reliability analyses have been prepared for each sensor. The results of these studies are summarized in sec. 6 of this volume and in detail in vol. II, secs. 9 and 10. On the basis of these studies, the following general conclusions and recommendations are drawn:

1) In most cases, the electro-optical sensor performance requirements can be met with existing designs. For those sensors requiring development (precision Canopus tracker and planet tracker), the requirements can be met without requiring significant advances in the present state-of-the-art.

2) Gimbaled planet tracker and precision Canopus trackers are preferable to body-fixed sensors to meet the high-accuracy requirements for Mars approach guidance.

3) It is recommended that the existing sensor electronics be redesigned and repackaged to provide a simple and reliable interface with the digital computer and to integrate the electronics into a minimum number of separate packages.

For the two inertial reference units studied, representing two different ranges of available performance capabilities, the mission requirements can be met in all cases by the less accurate unit. Either unit may be used interchangeably in the modular design.

2.2.5 Performance Capabilities and Limitations

Parametric trajectory accuracy analyses for the radio/optical/strapdown inertial guidance system have been conducted for the five missions under consideration. The results of these analyses are summarized in sec. 5 of this volume and presented in detail in vol. II, sec. 4. On the basis of these parametric performance analyses and the mission accuracy requirements (see sec. 3), performance requirements have been established for each of the electro-optical sensors and for the inertial unit. The sensor accuracy requirements are summarized in sec. 7, Tables 7-II and 7-IV. The trajectory accuracy and fuel required for correction of the guidance, navigation, and control errors are summarized in sec. 6, Table 6-III. On the basis of these system performance studies, the following conclusions are drawn:

1) The mission performance requirements (summarized in sec. 3) are met for the five missions under consideration by the recommended modular design.

2) For the Mars Orbiter mission, data from a planetary approach sensor, in conjunction with data from the sun and Canopus sensors, can be utilized by ground-based stations to improve the quality of the determination of the spacecraft approach orbit to Mars. However, for the requirements projected for the Voyager mission, this improvement in accuracy is not essential. The planetary approach sensor has been included conditionally in the system configuration so that the implications on preliminary modular design can be investigated for applications to possible future missions with high-accuracy requirements.

The specific guidance and control system configurations recommended in this study for each of the missions represent a minimum assemblage of hardware necessary to meet the functional and performance requirements established by the mission. Reliability, weight, and power estimates are summarized in subsec. 6.6 of this volume.

The following conclusions are drawn with respect to the system reliability:

3) For the earth orbit and lunar missions, the guidance and control system reliability is sufficient to meet reasonable mission reliability goals without the use of redundant elements. For the longer interplanetary missions, an unacceptably low reliability results if all elements of the system are operated throughout the mission. Selective incorporation of redundant elements and partial shutdown during the long interplanetary cruise phases will allow the system to meet reasonable mission reliability goals.

4) The most unreliable elements of the guidance and control system are the inertial reference unit, the digital computer, and the control system electronics. An attitude hold mode utilizing the Canopus sensor and sun sensor with simple analog electronics is incorporated in the system design permitting the most unreliable elements to be switched off during the interplanetary cruise phases.

2.3 RADIO GUIDANCE UTILIZATION

One objective of the Radio/Optical/Strapdown Inertial Guidance Study was to determine the role of radio command in the guidance of unmanned launch vehicles employing the advanced kick stage. The principal results and conclusions of this study, presented in sec. 6, Ref. 2-1, are summarized here.

2.3.1 Assumptions and Ground Rules

The assumptions and ground rules used in conducting this study are as follows:

a) Only existing NASA and DOD radio tracking systems are considered, i.e., no new systems are postulated nor has relocation of existing equipment been considered. The tracking systems considered are those shown in Table 2-II.

b) Those tracking systems that cannot be used for near real-time trajectory or orbit determination without major additions of equipment such as ground links, ground computational facilities, ground/vehicle data links, etc., are not considered. Generally, this eliminates the range instrumentation systems such as MISTRAM,* AZUZA, UDOP, GLOTRAC, etc.

TABLE 2-II

RADIO TRACKING SYSTEMS CONSIDERED IN STUDY

System	Location
DOD Systems • GE Mod III • BTL	Eastern and Western Test Ranges (Cape Kennedy and Vandenberg AFB)
NASA Systems • STADAN • C-Band radars† • Unified S-Band System (USBS) • DSIF	World-wide deployment. See Ref. 2-1 for station, locations

†DOD C-Band radars are included.

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*MISTRAM has a limited real-time capability (it is used for range safety). Its uncertain future makes it questionable for this application.

Descriptions of the BTL and GE Mod III radio/inertial guidance systems, the NASA STADAN system, the C-Band and S-Band (USBS) trackers, and the DSIF are given in subsec. 6.3 of Ref. 2-1. Further information on the NASA systems is given in Ref. 2-2 through 2-9. Error models for these systems (with the exception of STADAN) are given in Ref. 2-1, subsec. 6.4. The error model for the DSIF tracking system, which plays an essential role in the interplanetary missions, is given in par. 2.2.4.4.

2.3.2 Radio Guidance System Concepts and Tradeoffs

The methods of implementing radio command guidance which were considered in this study are:

a) A ground-based computer, receiving information from a radar or radar net during plwered flight, computes engine on-off commands and transmits turning rate commands to an onboard attitude control system.

   An example of this type of system is the radio-guided Atlas (GE Mod III System). It requires a minimum of onboard inertial equipment but is satisfactory only for near-earth operations because of transit time delays. It also has the disadvantage of constraining the maneuver times because of incomplete coverage. A second example of such a system is the BTL radio/inertial system used for Thor/Delta and other vehicles. In both systems, a radar is used to track during powered flight and a filter is used to estimate the position, velocity, and acceleration components. Because the acceleration components are estimated by the filter, only a minimum of inertial equipment (an autopilot) is required. The system errors are the result of an optimum weighting between the radar noise and the vehicle uncertainties (thrust, I_{sp}, mass) and autopilot gyro drifts.

b) A ground-based computer, receiving information from a radar net during free flight, computes the time of initiation, direction, and magnitude of a desired velocity increment. The required onboard equipment includes a sequencer to start and control the burn, an attitude reference system including optical alignment devices, and an integrating accelerometer. This type of system was used for the Ranger/Mariner midcourse corrections and is satisfactory mainly for small burns.

The errors in this type of system are in determining the desired velocity increment and in the execution of the burn. The errors in determining the desired velocity increment are the result of errors introduced during free-flight tracking by radar noise and biases. Execution errors are the result of inertial and optical instrument errors and vehicle dispersions. The vehicle dispersions cause errors in three ways:

1) Thrust misalignments and center-of-gravity offsets introduce directional errors. It is possible to use accelerometers to sense and correct these errors.

2) Thrust, weight, and I_{sp} dispersions cause the burnout position to deviate from nominal. Without onboard computing capability, the velocity increment cannot be modified to compensate for these errors.

c) Ground tracking during free flight is used to provide a position and velocity estimate which is used to update a complete inertial guidance system onboard the spacecraft. The Apollo mission will utilize this type of guidance.

The errors in this system are caused by radar noise and biases during free-flight tracking and inertial and optical instrument errors as well as thrust tailoff impulse.

d) Inertial guidance is used without radio aid. Although this type of guidance is conceivable for a synchronous satellite mission, it is totally unfeasible for a lunar or interplanetary mission unless some sort of terminal navigation sensor is used. Depending on the mission requirements, this may be beyond the current state-of-the-art.

The candidate radio/inertial systems considered in this study are shown in Table 2-III and include systems of all four types.

2.3.3 General Conclusions and Recommendations

The following general conclusions result from this study:

a) Use of the C-Band radars is limited to low-earth orbit tracking only. Station locations, coverage, data communication constraints, and system accuracy limitations are such as to eliminate these systems from consideration as useful radio guidance systems for the missions considered. However, tracking

TABLE 2-III

CANDIDATE SYSTEMS - RADIO/INERTIAL GUIDANCE SYSTEM COMBINATIONS

Type	Ground Based System	Vehicle Subsystems	Tracking System	Limitations
1	Tracking radar computer, data link to vehicle Cutoff and steering commands generated in ground computer on the basis of tracking information and transmitted to vehicle	Transponder/Decoder Autopilot Gyros torqued by turning rate commands	GE-Mod III (Atlas) BTL (Titan I, Thor Delta)	Existing systems limited to launch guidance only Minimum elevation angle 5 deg
2	Tracking radar(s), ground communication net, computer, data link to vehicle	Partial inertial system (e. g. Ranger/Mariner) • Attitude ref. (optically aided) • Single accel.	C-Band radars	• USBS limited to near earth and lunar missions • Appreciable time required for gathering data and computing trajectory
3	Tracking data used to compute orbit. Orbital data transmitted to vehicle over data link	Complete inertial system • Attitude ref. (optically aided) • 3-axis IRU • Computer	S-Band radars (USBS) DSIF	• C-Band systems limited to low altitude earth orbits; coverage limited
4	None	Complete inertial system (IRU and computer) with optical aids (as required)	None	Adequate for synchronous orbit injection, lunar/planetary orbit injection

and orbit determination of spacecraft in low-altitude earth parking orbits is possible to reasonable accuacies (as was done on the Gemini program).

b) The GE Mod III and BTL radio/inertial systems may be used for accurate guidance during the launch phase from both ETR and WTR. These systems are currently in use for Atlas/Agena and Thor/Delta launches. A limitation is reached when the elevation angle of the vehicle, as seen from the radar site, drops below 5°. This condition is reached prior to orbit insertion for most vehicles employing upper stages such as Centaur, Agena, and Delta (final stage). Nevertheless, it is possible to use these systems to guide the lower stages of certain multistage vehicles and to "turn over" the guidance to the onboard systems at the appropriate time during the mission.

c) The use of the NASA STADAN net is useful for long-term tracking of spacecraft in earth orbit. Its use is suggested for the synchronous earth orbit mission (after final orbit insertion) for long-term orbit determination and station-keeping. The vehicle equipment required is normally associated with the mission payload and not considered to be part of the launch vehicle guidance.

d) The NASA Unified S-Band and DSIF nets provide excellent coverage and orbit determination capabilities for the lunar and interplanetary missions. These systems require extensive ground communications and computational facilities. The USBS is generally limited to near-earth and lunar missions. The DSIF net extends this capability to interplanetary distances.

e) The use of the DSIF for tracking and orbit determination is virtually a necessity for the interplanetary missions. Although completely autonomous onboard optical/inertial systems may be conceived for these missions, the required performance is considerably beyond the present state-of-the-art for most missions. An accurate onboard system is required, in any case, for controlling accurately powered maneuvers such as midcourse corrections and orbit insertion maneuvers.

2.3.4 Limitations, Constraints, and Performance Capabilities

Radio guidance performance capabilities, limitations, and constraints for the earth orbit, lunar, and interplanetary missions are summarized below for each significant mission phase.

2.3.4.1 Use of Radio Guidance - Launch Through Parking Orbit or Interplanetary Orbit Insertion

Radio guidance is currently in use for several NASA launch vehicles (Atlas/Agena, Thor/Delta, Titan II/Gemini) and AF launch vehicles, (Titan III, Atlas/Agena, Thor/Delta). Launch phase radio guidance is provided for these vehicles using either the GE Mod III or BTL radio/inertial guidance systems. In all cases, the tracking radar is located in the vicinity of the launch site and tracks the vehicle to the lower elevation angle limit (5° to 10°, depending on the mission accuracy requirements). By suitably shaping the launch trajectory to maintain acceptable elevation and vehicle antenna look-angles, accurate guidance can be provided through the first two, and portions of the third, stages of powered flight. For Altas/Agena, guidance is assumed by a simple onboard inertial system (attitude reference, programmer, horizon scanner, and a single axially mounted accelerometer for thrust cutoff) during the Agena burn. The radio guidance serves to initialize the inertial system.

A number of difficulties are encountered in extending the use of radio guidance to vehicles employing high performance upper stages (Atlas/Centaur) or requiring additional stages to meet the requirements of higher energy missions. As indicated below and in sec. 6, Ref. 2-1, the best available tracking radars suitably located at downrange sites will meet the launch-phase guidance requirements for many lunar and interplanetary missions. However, there are severe siting and related problems such as acquisition and vehicle antenna coverage. Some payload (weight) penalties and launch azimuth (and consequently launch window) constraints are incurred due to tracking system geometrical constraints. Trajectories are, in general, limited to direct ascent types. The whole approach of using radio guidance with parking orbit trajectories appears impractical. (See par. 6.5.1.1, Ref. 2-1, for further discussion.)

The analysis of radio guidance feasibility and performance during the launch through injection phases has been based on a lunar mission, and an Atlas/Centaur trajectory has been assumed. Performance results are presented in Ref. 2-1, par. 6.5.1.2. The performance criterion used is the midcourse ΔV correction required to correct the miss and time of flight errors at the moon due to the launch guidance errors. Typical figure of merit (FOM) values for this mission are 10 m/sec (1σ).

2.3.4.2 Orbit Determination Accuracy During Earth-Orbit Coast

Numerous studies have been made of orbit determination accuracies for spacecraft in low- and high-altitude earth orbits in support of Mercury, Gemini, Apollo, and other NASA and DOD space programs. Some results from these studies that are particularly pertinent to the present study are summarized in Ref. 2-1, par. 6.5.2. Use of all available NASA C-Band and USBS tracking stations is assumed, as is the availability of appropriate computing facilities for near real-time orbit computation.

Predicted orbit determination accuracies for a vehicle in a low-altitude (185 km) earth orbit are given in Ref. 2-5. The results show rapid degradation in the vehicle velocity uncertainties when continuous coverage tracking is not available. This data indicates the need for multiple stations to achieve reasonable orbit determination accuracies.

The conclusion is drawn from these results that the use of radio guidance during low altitude parking orbit coast phases is not practical for the Missions and vehicles covered in this study. This is due to a combination of tracking system coverage limitations, tracking system performance limitations, and time delays inherent in gathering the data, transmitting it to a central computing facility, reducing the data, computing vehicle commands, and transmitting these commands via data link to the orbiting vehicle.

For the synchronous orbit mission, it is shown in sec. 7, Ref. 2-1, that a navigation update* is required prior to synchronous orbit injection for missions that involve long parking orbit coast times. This correction can be made by either of two methods:

*This is in addition to the attitude updates required prior to the transfer burn and final orbit insertion. The errors to be corrected are primarily the accumulated position errors.

a) Use of radio tracking during the transfer orbit coast to determine the position error. The major part of the error can be removed by proper adjustment of the time of initiation of the final orbit insertion burn.

b) Use of an onboard electro-optical sensor (e.g., a sun sensor to establish a "line-of-position" fix at some point during the transfer orbit coast. The position error is removed as in a) above.

The feasibility of method a) depends on the availability of suitably located tracking stations. The desired location depends on the longitude of the satellite desired after injection into the final synchronous orbit. Although it may be possible to select suitable tracking stations for most final longitudes of interest, some operational and trajectory constraints are evident.* The use of the second method, which can be implemented entirely within the onboard system, appears very attractive.

For tracking a spacecraft after injection into the final synchronous orbit, ground based tracking is somewhat more useful. Such a capability is useful for long-time stationkeeping which requires periodic orbit prediction and adjustment that may be easily implemented with either the S-Band systems or the NASA STADAN net. The latter system is recommended for this purpose.

2.3.4.3 Orbit Determination Accuracy During Translunar Trajectory Phases Using the S-Band Tracking Systems

Exhaustive studies have been made of orbit determination accuracies for lunar missions in support of the Apollo (Ref. 2-10), Lunar Orbiter, and other programs. Similar but less exhaustive studies have been made for various interplanetary missions (Refs. 2-11 and 2-12). Some results from these studies are summarized here that are pertinent to the present study. Additional study results for the Mars mission are presented in sec. 9, Ref. 2-1 and sec. 4, vol. II, and summarized in subsec. 5.7 of this volume.

*It is also possible to use different modes of ascent from the one studied here. One commonly used technique is to inject the satellite into an equatorial orbit whose period is substantially different from 24 hr and let the satellite "drift" to the required longitude, at which point the orbital period is corrected.

The results of tracking accuracy studies are normally computed in the form of state-vector uncertainties as a function of time from injection. The quantities used to represent the uncertainties are the square root of the sum of the variances of the three position and velocity components.

The results presented in Ref. 2-10 and summarized in sec. 6, Ref. 2-1, indicate that launch azimuth, earth orbital coast type, flight time, and launch data have effects on DSIF tracking during the early portion of the flight due to their effects on coverage. In the latter portion of the trajectory, the accumulated accuracy of DSIF tracking is nearly independent of the trajectory. Flight time is the only trajectory parameter with a noticeable effect on the latter portion of the trajectory. C-Band radar is found to be useful in reducing uncertainties in the early part of the flight, but it is limited to tracking the first 1.5 hr of the trajectory. The addition of range information to this network gives a marked improvement in tracking accuracy.

Table 2-IV presents some typical results of position and velocity uncertainties at encounter for various tracking system configurations with and without the simulation of midcourse correction effects.

TABLE 2-IV

TYPICAL TRANSLUNAR TRAJECTORY DETERMINATION ACCURACIES

	Midcourse Correction Effects Not Included		Midcourse Correction Effects Included	
Data Type	1σ Position Uncertainty (km)	1σ Velocity Uncertainty (m/sec)	1σ Position Uncertainty (km)	1σ Velocity Uncertainty (m/sec)
DSIF (range, range rate, angle data)	0.1	0.06	0.8	0.46
DSIF (no range)	2.1	1.5	3.7	2.9
C-Band radar	1.9	0.37	---	---

The USBS/DSIF network assumed to be tracking the spacecraft during the translunar trajectory consists of Goldstone, Canberra, and Madrid. Table 6-VII, Ref. 2-1, lists the locations of these stations.

Additional results indicating the tracking capability during the translunar trajectory with earth-based radar are presented in par. 6. 5. 3. 4, Ref. 2-1. Certain generalizations can be made, keeping in mind the assumptions of this study.

The position and velocity uncertainties associated with radar tracking only may be characterized by the following properties:

a) Sensitivity over the early portion of the trajectory to launch azimuth, type of coast, flight time, and date of launch, due to changes in tracking coverage

b) Insensitivity to the trajectory parameters over the latter portion of the trajectory

c) Large uncertainties in the downrange direction (measured in orbit plane coordinates)

d) Sudden drops in the overall uncertainties at the start of periods of simultaneous or near simultaneous tracking by two stations when range data are used.

In general, it can be said that DSIF tracking is greatly improved by the addition of range information, particularly if simultaneous or near simultaneous tracking by two stations is possible.

2. 3. 4. 4 Interplanetary Orbit Determination Accuracy Using DSIF

For purposes of this study a DSIF error model has been establised based on the guaranteed and probable range-rate measurement errors given in Ref. 2-9.

For purposes of this study, a conservative value intermediate between the guaranteed and probable accuracies for the 1970 time period has been selected (essentially equivalent to the present probable accuracy). In addition, a range-rate bias error is assumed, uncorrelated from station to station. These errors are shown in Table 2-V.

The use of DSIF for tracking and orbit determination is virtually a necessity for the interplanetary missions considered in this study. The results presented in sec. 9, Ref. 2-1, for the Mars Orbiter mission show

TABLE 2-V

DSIF RANGE RATE ERRORS ASSUMED
FOR ANALYSIS PURPOSES

Error Source	RMS Error
Uncorrelated noise on doppler rate	0.732×10^{-2} m/sec (equivalent to 0.12 ft/sec per 1 sec sample, 25 measurements averaged) (also equivalent to 0.0006 m/sec uncorrelated rms error a 1 sample/min)
Range-rate bias	10^{-2} m/sec (0.0328 ft/sec)

that a completely autonomous onboard optical/inertial system cannot meet the desired mission accuracy requirements within the present (or near future) state-of-the-art. However, use of the onboard optical/inertial system in conjunction with DSIF is attractive both in terms of accuracy and accuracy and operational utility. In this mode of operation, DSIF is used as the primary source of accurate position and velocity data (with respect to the earth) and the onboard system is used to accurately control the mid-course, orbit insertion, and orbit trim maneuvers. Use of onboard sensors is also helpful in determining the spacecraft orbit relative to a planet whose position with respect to the earth is uncertain to a significant degree. See sec. 4 of vol. II for a more detailed discussion.

The orbit determination accuracies attainable with DSIF depend on the mission trajectory and will change significantly throughout the mission. Table 2-VI presents some typical present and future capabilities for the Mars mission. A comparison is also made with the expected errors at encounter in the absence of tracking data for a typical launch injection guidance error of 10 m/sec.

2.3.4.5 Lunar and Planetary Orbit Determination Accuracy

Estimates of the accuracy of lunar orbit determination from unified S-Band (USBS) and DSIF tracking data have recently been revised based on postflight analysis of the Lunar Orbiter 3 tracking data. Subsec. 4.7 of vol. II contains a discussion of the estimated tracking accuracies achievable for lunar orbit. Table 2-VII summarizes the present capability based on analysis of LO3 data.

Mars orbit determination capabilieies using DSIF doppler data have been analyzed as part of this study. The results are summarized in subsec. 5.8 of this volume.

TABLE 2-VI

TYPICAL TRAJECTORY DETERMINATION ACCURACIES FOR A MARS MISSION

	Error At Encounter
Launch Injection Guidance Only	
10 m/sec	90,000 - 200,000 km
Earth Based Tracking Using DSIF	
Present (Mariner 4 Results)	
• 5 days after injection	2400 km
• All data including post encounter tracking	500 km
Future (1971)	
• Injection - 5 days	1000 km
• 5 to 120 days	150 km
• After 120 days	100 km

TABLE 2-VII

ESTIMATED LUNAR ORBIT NAVIGATION UNCERTAINTIES IN RTN COORDINATES (SEE PAR. 1.4.5)

Error	At Time of Tracking	Propagated for Two Orbits (no tracking)
σ_R	1000 ft	2600 ft
σ_T	3000 ft	8544 ft
σ_N	300 ft	1044 ft
σ_R	7.3 ft/sec	7.3 ft/sec
σ_T	2.2 ft/sec	2.2 ft/sec
σ_N	9.2 ft/sec	9.3 ft/sec

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3. SUMMARY OF MISSION CHARACTERISTICS, GUIDANCE AND CONTROL SYSTEM OPERATIONAL SEQUENCES, AND PERFORMANCE REQUIREMENTS

3.1 INTRODUCTION

This section presents a summary of the mission characteristics and requirements, and the guidance system functional and performance requirements derived from them. Mission characteristics and guidance system performance for the earth-synchronous orbit mission and the lunar orbiter mission are essentially the same as those developed under the Task I and II effort, and documented in Ref. 3-1. Under the Task III study effort, the Mars orbiter and solar probe (Jupiter assist) missions have been revised and the low-attitude earth polar-orbit mission added.

The mission and launch vehicle characteristics, trajectories, and mission performance (accuracy) requirements are summarized under each mission heading. Guidance system functional requirements and operating sequences derived from the mission requirements, vehicle characteristics, and guidance equipment capabilities are specified. Equipment configurations and functional interconnections are presented for each of the missions in sec. 4.

3.2 EARTH LOW-ALTITUDE POLAR-ORBIT MISSION

The earth low-altitude polar-orbit mission typifies one that might be used for earth resources studies and was included in the mission repertory to provide broad coverage of the spectrum of possible unmanned space missions. This study assumed that the orbiting satellite payload is capable of correcting for orbit-insertion errors. Typically, this ΔV capability can be on the order of 10 to 20 m/sec. The modular guidance system must then provide the guidance function from launch through orbit insertion with accuracy sufficient to ensure that the payload ΔV capability is not exceeded.

For this mission, the Atlas/Burner II launch vehicle is assumed to be launched from the Western Test Range (WTR). The Atlas stages inject the Burner II/payload combination into a coast up to the apogee altitude of

927 km. At that altitude the Burner II provides the velocity increment for circularizing the payload orbit. The actual sequence of events is summarized in Table 3-I. Basic data used to define the launch and injection trajectory and this sequence of events were obtained from Refs. 3-2 through 3-5.

TABLE 3-I

ATLAS SLV-3A/BURNER II SEQUENCE OF EVENTS

Event	Description	Time (sec from Liftoff)
T_{LO}	Liftoff	0.0
BECO	Booster engine cutoff (sustainer operation)	148.4
JBP	Jettison booster package and shroud	151.4
SECO	Sustainer engine cutoff	361.4
VECO	Vernier engine cutoff (begin coast to apogee)	381.1
B2IG	Burner II ignition	1107.1
B2BO	Burner II burnout (circular orbit injection)	1153.1

The characteristics of the actual orbit obtained from the TRW/N-Stage program include the following:

Injected weight	2513 lb
Inclination	99°
Apogee/perigee	954/900 km
Eccentricity	0.0037
Orbital period	103.54 min

This is not a perfectly circular orbit. Since the above orbit was adequate for error analysis purposes, further iterations of the N-Stage program to achieve a more circular orbit were not attempted.

3.3 SYNCHRONOUS EARTH SATELLITE MISSION

For this mission, the launch vehicle is assumed to be the Atlas SLV3X-Centaur* with a variety of communication and meteorological satellites as the payload. It is assumed that the satellite payload itself has the capability of providing a ΔV for final orbit trim and stationkeeping. The ultimate functional and performance requirements imposed on the kick stage for this mission are to place a payload into a near synchronous earth orbit, at the desired longitude, with sufficient precision that final orbit trim corrections can be performed utilizing the limited propulsion capability of the payload. The kick stage guidance system accuracy requirements may be conveniently stated in terms of the payload ΔV required to correct the residual errors after final injection. Reasonable values lie in the range of 15 to 30 m/sec.

For purposes of this study, it is assumed that the ROI guidance system provides the complete guidance and control of the launch vehicle from liftoff through parking orbit insertion, transfer injection, and synchronous orbit injection. Two extremes of ascent trajectories have been considered. In the first, the kick stage is injected into the transfer trajectory to synchronous altitude from a 185-km "parking orbit" at the first equatorial crossing from launch. In the second, the kick stage/payload may remain in the 185-km parking orbit for as long as 12 hr before transfer ignition. These are the extremes of the parking orbit coast period required to reach any desired final longitude for this mode of ascent.

*The payload and coast duration capabilities of this vehicle are severely limited for this mission using the existing Centaur vehicle. For the purposes of this study, these problems are ignored. It is assumed that the Centaur vehicle may be modified to increase the payload capability, to extend the permissible coast duration, and to permit three-burn operation. Another alternative, providing a large increase in payload capability, is to add an upper stage (such as HEUS) to the vehicle. The guidance requirements are not expected to be significantly different for either vehicle concept.

3.3.1 Mission Characteristics

For the purposes of this study, the major events of the synchronous mission developed for the Atlas/Centaur (AC-8 configuration) have been adopted and modified. Following liftoff from the Atlantic Missile Range, a roll is introduced in the launch vehicle to obtain a launch azimuth of 90 deg. The Atlas booster is then controlled up to its cutoff (BECO) by a predetermined booster pitch program.

Injection into the parking orbit is accomplished by using two constant pitch rates selected to achieve the altitude and flight path angle for injection into the 185-km parking orbit. The first pitch rate occurs during the Atlas sustainer flight, lasting for 10 sec after initiation of that phase, while the second rate occurs during the Centaur powered phase. After injection into the parking orbit, the Centaur coasts to the vicinity of the equator (first crossing) at which time the second burn (approximately 1.5 min) injects it into a Hohmann transfer ellipse. This burn is performed with a pitch rate that keeps the Centaur in a fixed attitude relative to the radius vector, and terminated on a predicted apogee altitude equal to that of the required synchronous circular orbit. During the coast in the Hohmann transfer, approximately 5 hr, the Centaur maintains a fixed inertial attitude.

Optimally, minimum energy requirements suggest dividing the orbit inclination plane change between perigee and apogee. For launch from AMR at 90-deg launch azimuth, the orbit inclination is 28.5 deg; approximately 2 deg should be removed at perigee and the remaining 26.5 deg at apogee. For this study, the gains from pursuing this approach do not overcome the complexities introduced. Therefore, the method adopted for the third Centaur burn at apogee was to perform the total orbit plane change simultaneously with injection into the synchronous orbit.

Just prior to reaching apogee, instantaneous yaw and pitch attitude maneuvers were performed to establish an initial attitude for the final burn (approximately 39 sec) such that the Centaur would achieve the correct synchronous orbit. Characteristics of the actual synchronous orbit obtained are:

•	Altitude	35,850 km (19,326.5 n.mi.)
•	Longitude	102.7 deg
•	Velocity magnitude	3.08 km/sec (10,087.3 ft/sec)
•	Eccentricity	0
•	Inclination	0 deg
•	Period	1436.1 min

After injection into the circular synchronous orbit, the payload separates from the Centaur. Any errors in the resulting spacecraft orbit are then corrected by the spacecraft itself.

Developing the nominal trajectory presented above was contingent upon making the following simplifying assumptions:

a) A mission of this type requires a three-burn capability from the Centaur. Since presently only a two-burn capability is available,* the detailed sequence of events of the second burn was duplicated for a third burn.

b) Payload maximization could be obtained by optimizing several trajectory parameters such as launch azimuth, plane change philosophy, parking orbit altitude, vehicle attitude history, etc. However, for this guidance study, the exact maximum payload weight is irrelevant to the guidance scheme adopted. Hence, no payload maximization analysis was performed.

c) Positioning a 24-hour synchronous spacecraft above a specified longitude may also be accomplished by injecting into an orbit offset from the required circular synchronous orbit. A drift rate results which allows the spacecraft to change its longitude. This drift rate is

*A two-burn (Centaur Stage) mission profile is also possible, using the technique as discussed under c). Although the three-burn capability and the extended coast capability required for either mission profile is not in the present Centaur design, these capabilities could be provided by an improved Centaur stage or an alternate stage having similar capabilities. It is beyond the scope of this study to assess the technical feasibility of these design changes.

then removed, and the final orbital corrections are made when the required longitude is reached. Since these corrections would be executed by the spacecraft and not the launch vehicle, guidance techniques for the Centaur would not be affected if such considerations were incorporated into this analysis. Consequently, the spacecraft was targeted directly into the 24-hr synchronous equatorial orbit, thus neglecting offset drift-orbit considerations.

d) An eight-orbit phasing coast in a 185-km parking orbit is simulated for certain runs by the analytical propagation of errors in the error analysis program (see sec. 7, Ref. 3-1). The remarks in b) above concerning Centaur capabilities apply here as well. The event times for the synchronous orbit missions are given in Ref. 3-1, Tables 2-II and 2-III, for cases without and with an eight-orbit phasing coast, respectively.

3.3.2 Guidance System Operational Sequence

The guidance system operational sequence during each of the mission phases is summarized below:

a) Launch and boost to ~ 185-km parking orbit:* The strapdown inertial guidance subsystem is presumed to be providing the guidance function for this phase.

b) Coast in parking orbit for a period t, with t depending on desired longitudinal positioning of satellite (15 min < t < 12 hr): During the coast period, the inertial guidance subsystem is required only to provide vehicle attitude control reference. The exact attitude profile to be followed during the coast phase will depend on the mechanization concept developed; however, at transfer ignition (at equatorial crossing) the kick stage attitude must be at that thrusting attitude required to place the kick stage/payload into the desired transfer

*This is a typical value assumed for this study and represents a reasonable lower limit for this type of mission. The parking orbit altitude is chosen as low as possible to maximize the injected payload weight. However, below about 185 km, drag effects limit the orbital lifetime of the vehicle.

orbit. The attitude control during the period immediately prior to transfer ignition might be inertial only or optically aided inertial using earth (horizon) and sun sensors.*

c) Transfer burn to apogee: This phase will be controlled autonomously by the strapdown inertial guidance subsystem.

d) Transfer coast: During the approximately 5-1/4 hr coast in the Hohmann transfer to the apogee at nominal synchronous altitude, the inertial guidance subsystem can again be relegated to the role of an attitude reference set.*

e) Apogee burn: The apogee burn is designed to circularize the orbit at synchronous altitude and is controlled by the strapdown inertial guidance subsystem. The use of the kick stage is presumably terminated at this time and the payload is separated from the kick stage.

3.3.3 Guidance System Performance Requirements

Because of (1) imperfect tracking or navigation during the transfer coast and (2) thrusting attitude and ΔV errors of the kick stage at apogee burn, the payload orbit will be imperfect in several respects:

a) The orbit is, in general, elliptical.

b) The orbital inclination is, in general, not zero.

c) The longitude of the subsatellite point is, in general, not the desired longitude.

The capability of the payload propulsion to correct for these errors dictates the final accuracy requirements of the kick stage apogee burn. Subsec. 2.3 of Ref. 3-1 analyzes the relationship of trajectory errors to payload ΔV requirements.

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*Use of earth-based radio tracking systems for coast-phase orbit determination and updating of transfer and injection burn ignition time, and velocity vector increments is another possibility. (Subsec. 2.2 of Ref. 3-1 discusses the limitations of this technique.)

The results of this analysis are a set of nonlinear expressions relating position and velocity errors at injection to the ΔV required to correct the errors.* If ΔV_A represents the available payload propulsion capability, then the performance requirements for this mission may be stated as

$$\Delta V_{Total}\ (95\%) < \Delta V_A$$

where ΔV_{Total} is the value of ΔV required for 95% probability of successfully performing the correction. Reasonable values for ΔV_A lie in the range of 15 to 30 m/sec.

Results of a detailed performance analysis for this mission are presented in sec. 7, Ref. 3-1, and are summarized in sec. 5 of this volume.

3.4 MARS ORBITER MISSIONS

3.4.1 Mission Characteristics

Two 1975 Mars orbiter missions, corresponding to Type I and Type II transfers,** have been selected from the optimum 2-day launch periods identified during the Voyager Task D study (Ref. 3-6). These two types of trajectories were chosen to examine the sensitivity of the trajectory determination errors (and hence fuel required for corrective maneuvers) to guidance and control errors. For either type of trajectory, the basic mission phases listed below are identical:

a) Launch, parking orbit, and injection into interplanetary trajectory

b) Separation from booster and first-cruise phase

c) Midcourse execution

d) Subsequent cruise and midcourse corrections

e) Approach

*These expressions are given in par. 2.3.2 of Ref. 3-1.

**Type I transfers are defined as those in which the vehicle traces a central angle of less than 180° about the Sun between departure from the Earth and arrival at Mars. In Type II trajectories, the angle is greater than 180°. The two types are effectively noncontiguous: when the heliocentric central angle is very near 180°, the position of Mars out of the ecliptic causes the interplanetary trajectory to be highly inclined to the ecliptic, leading to excessive launch energy requirements.

f) Deboost velocity application (into 1,100 x 10,000-km orbit)

g) Doppler tracking in elliptic orbit

h) Transfer into 500-km altitude circular orbit

Within each launch period, the critical mission was identified as that Earth-Mars trajectory which requires the maximum short coast Earth parking orbit. Table 3-II summarizes the Saturn V launch vehicle characteristics used to compute these coast times. The basic booster data was obtained in Refs. 3-3 and 3-7. Specific launch sequence event times for the Type II transfer are summarized in Table 3-III.

Table 3-IV lists the pertinent trajectory characteristics of each critical mission; Figures 3-1 and 3-2 illustrate the heliocentric transfer geometry of each mission. Time histories of the following trajectory characteristics also are displayed in Figures 3-3 and 3-4 for the transit phase of each mission:

a) Sun-spacecraft distance

b) Sun-Mars distance

c) Spacecraft-Earth distance

d) Earth-Mars distance

e) Spacecraft-Mars distance

f) Sun-spacecraft-Earth angle

g) Sun-Mars-Earth angle

In addition, the Sun-Mars distance, Earth-Mars distance, and Sun-Mars-Earth angle plots have been extended to include the first 200 days of the orbiting phase of each mission.

The distances plotted in Figures 3-3 and 3-4 affect communications characteristics (spacecraft-earth distance) and relate to solar radiation and wind intensities (spacecraft-sun distance). The sun-spacecraft-earth angle is significant because of its effect on the transfer of attitude reference from earth to sun for the performance of midcourse maneuvers.

TABLE 3-II

SATURN V LAUNCH AND INJECTION TRAJECTORY CHARACTERISTICS

1957 MARS TRANSFERS			
Phase	Duration (min)	Angle (deg)	Altitude (n. mi.)
Type I Mars Transfer			
Total powered flight	17.28	49.80*	
Circular parking orbit	58.02	236.84*	100.0
Injection		8.0**	180.0
Type II Mars Transfer			
Total powered flight	17.20	49.20*	
Circular parking orbit	24.83	101.34*	100.0
Injection		8.0**	180.0

*Angle traversed, measured in earth-centered inertial coordinates.
**Flight path angle at injection, measured (+) above the local horizontal.

3.4.2 Guidance System Operational Sequence

The operation sequences for the Mars orbiter mission are assumed to be as outlined below:

a) Launch, parking orbit, and injection into interplanetary trajectory: The ROI system is used to inertially guide the Saturn V/payload from liftoff through injection.

b) Separation from booster and first cruise phase: The kick stage strapdown inertial subsystem is used to provide rate damping signals to stabilize the separation-induced tumbling transients. After the rate stabilization is accomplished, a celestial reference acquisition (Sun and Canopus) sequence is initiated. The Sun and Canopus trackers will be body fixed and will serve as the primary long-term inertial attitude references. After Sun/Canopus lock-on is achieved, the gyros may be turned off (except for heaters) until required for the midcourse reorientation maneuver.

TABLE 3-III

SATURN V SEQUENCE OF EVENTS
(TYPE II TRANSFER TRAJECTORY)

Event	Description	Time (sec from liftoff)
T_{LO}	Liftoff	0.0
IECO	S-IC inboard engine cutoff	154.6
OECO	S-IC outboard engine cutoff	158.6
T_{IG2}	S-II stage ignition	164.1
T_{JFI}	Jettison S-IC/S-II forward interstage	194.1
T_{JHS}	Jettison heat shroud	214.1
S2CO	S-II stage cutoff	538.1
T_{IG3}	S-IVB stage first ignition	543.6
T_{BO3}	S-IVB stage first cutoff (parking orbit injection)	686.2
T_{2IG3}	S-IVB stage second ignition	2184.9
S4CO	S-IVB stage final cutoff (transfer orbit injection)	2491.3

Deep Space Network (DSIF) tracking will be used during this cruise phase for orbit determination and to complete the first midcourse velocity correction required to reduce the effects of injection errors. The midcourse thrust vector pointing and magnitude commands and time of execution will be transmitted to the onboard guidance system for execution.

TABLE 3-IV

CHARACTERISTICS OF 1975 EARTH-MARS TRAJECTORIES

	Type I Transfer	Type II Transfer
Departure date	1975 September 19	1975 September 22
Arrival date	1976 May 1	1976 September 5
Time of flight, days	224.75	348.32
Departure asymptote (from earth)		
V_∞, km/sec	4.45	3.85
C_3, km^2/sec^2	19.76	14.83
Angle to equatorial plane, deg	50.12	5.13
Angle to sun-earth line, deg	248.94	255.14
Heliocentric orbit		
True anomaly at departure, deg	1.565	0.899
True anomaly at arrival, deg	7.204	-8.558
Heliocentric transfer angle, deg	150.68	203.32
Inclination to ecliptic, deg	3.751	2.083
Perihelion distance from sun, AU	1.003	1.003
Aphelion distance from sun, AU	1.705	1.675
Eccentricity	0.2594	0.2510
Approach asymptote (to Mars)		
V_∞, km/sec	3.09	2.80
Angle to plane of Mars' orbit, deg	-20.22	26.83
Angle to Mars-Sun line, deg	138.76	54.71

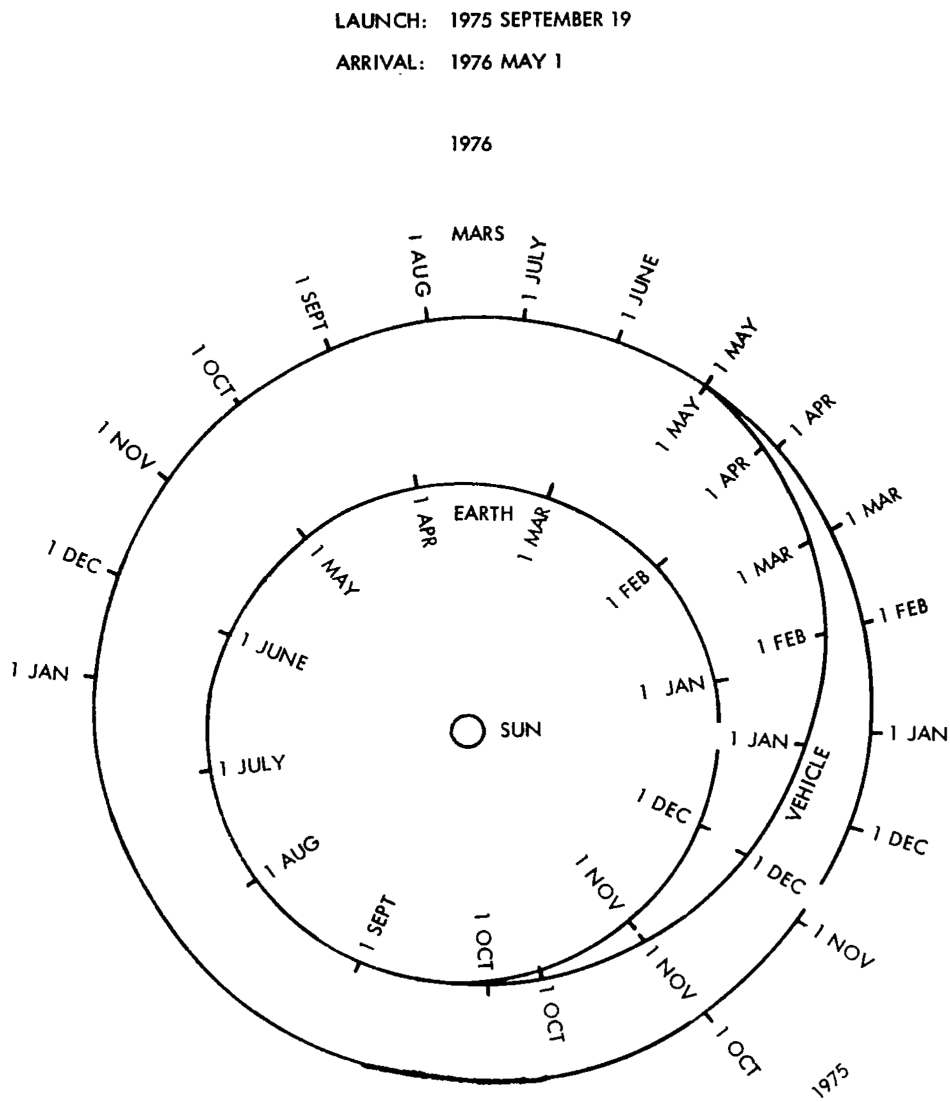

Figure 3-1. Ecliptic Projection of Sample 1975 Type I Mars Mission, Showing Relative Heliocentric Positions of Earth, Vehicle, and Mars

LAUNCH: 1975 SEPTEMBER 22

ARRIVAL: 1976 SEPTEMBER 5

1976

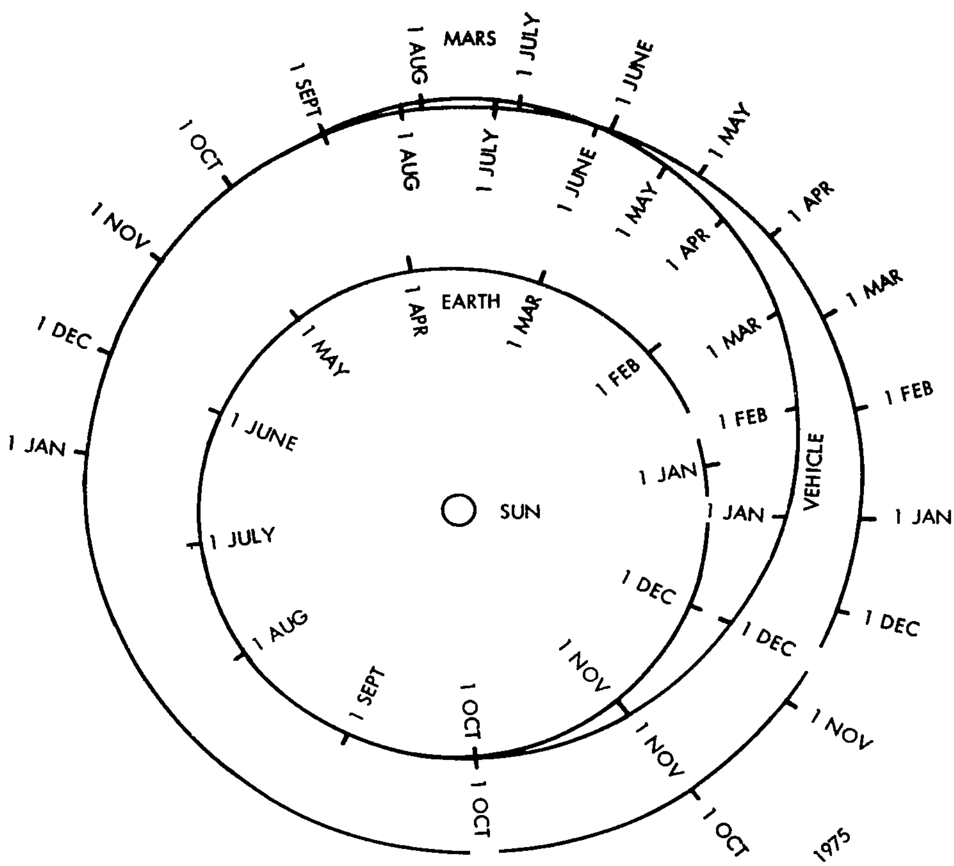

Figure 3-2. Ecliptic Projection of Sample 1975 Type II Mars Mission, Showing Relative Heliocentric Positions of Earth, Vehicle, and Mars

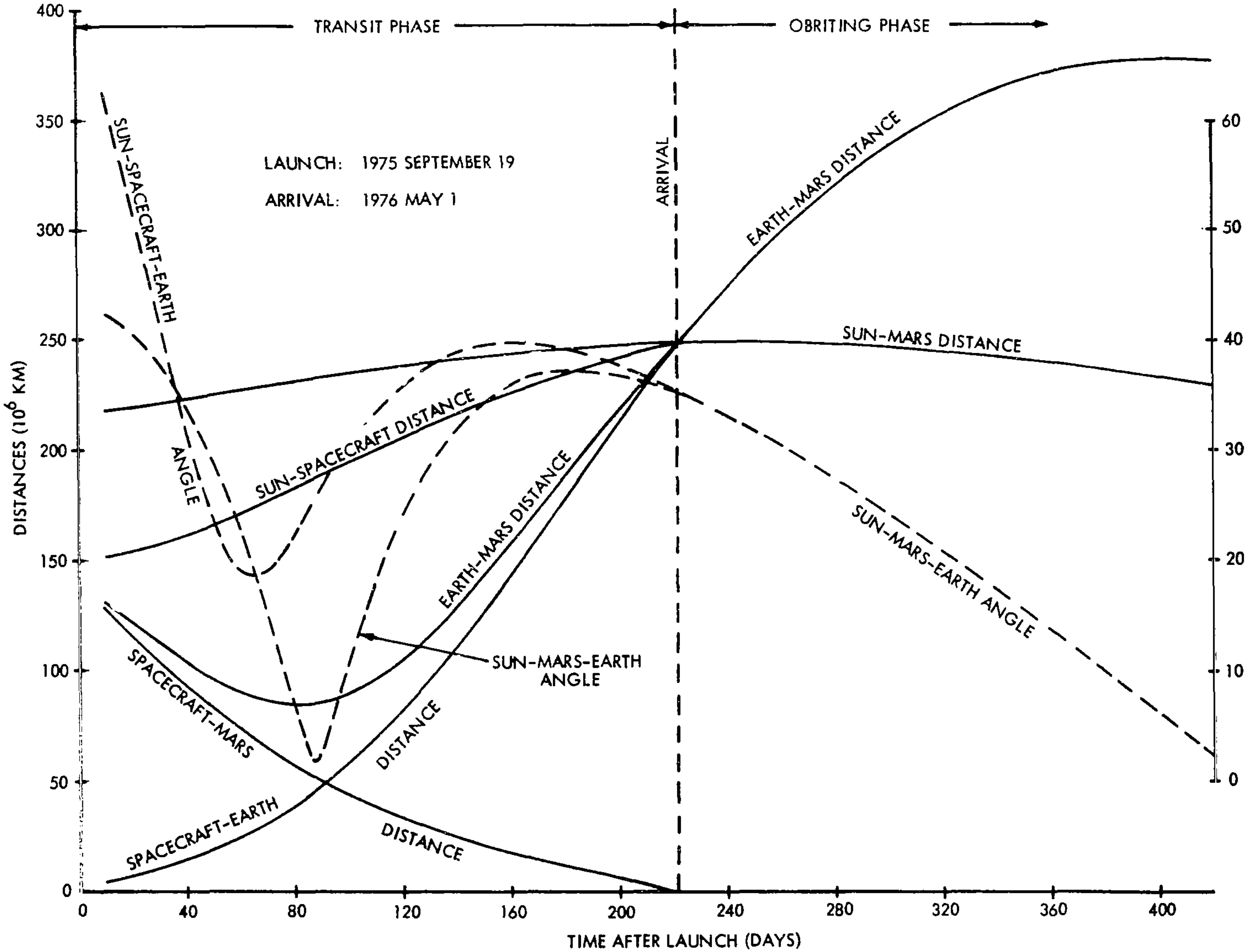

Figure 3-3. Time Histories of Heliocentric Orientation Angles and Distances for 1975 Type I Mars Mission

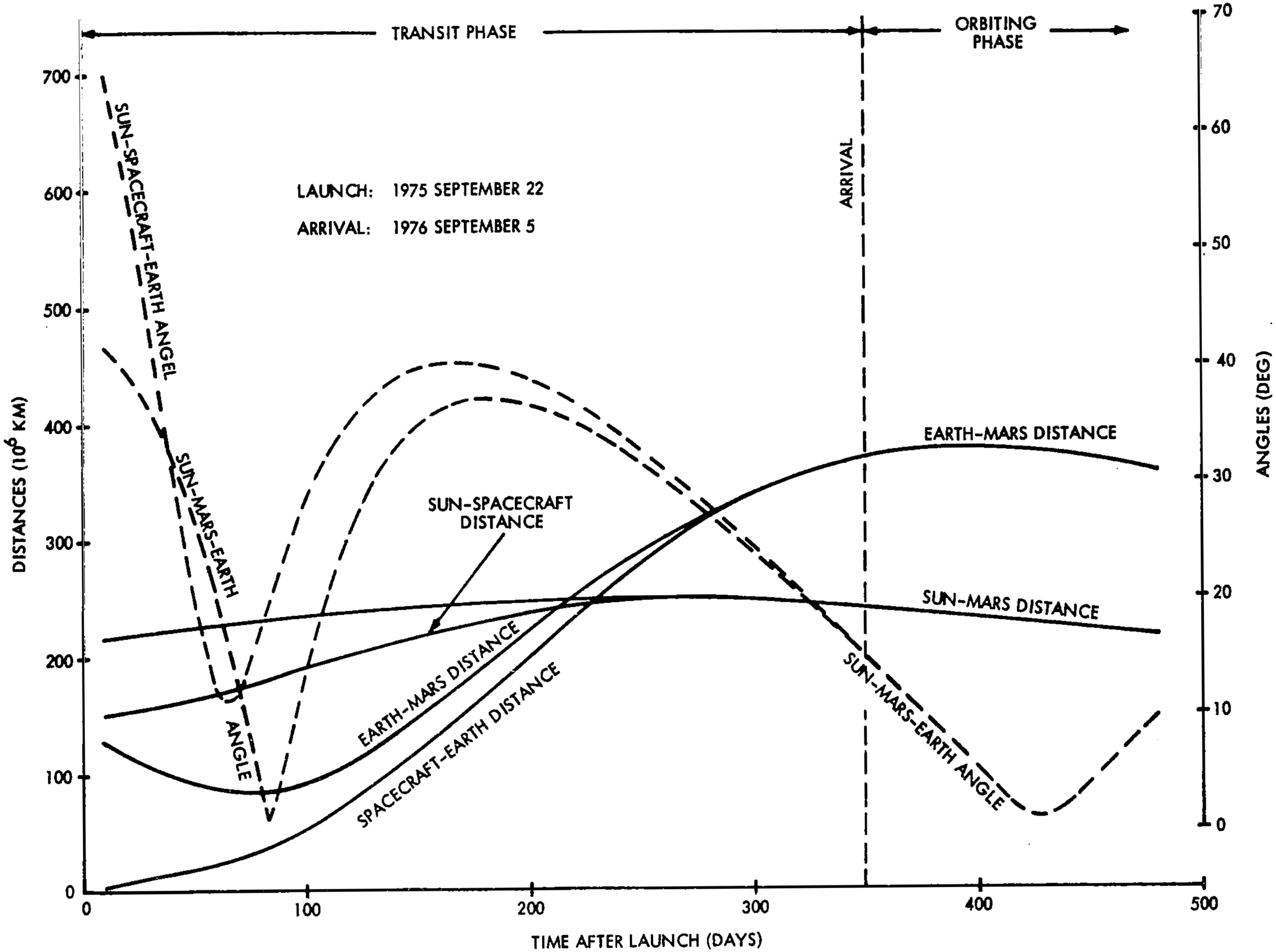

Figure 3-4. Time Histories of Heliocentric Orientation Angles and Distances for 1975 Type II Mars Mission

c) Midcourse execution: If the gyros were shutdown in the previous cruise phase, wheel power would be applied sufficiently early to ensure proper gyro operation during the following sequence. Ten to thirty min (the time will be dependent on available spacecraft slew rates and maximum required turn-through angles) prior to the time of execution of the midcourse correction burn, vehicle rotations will be commanded to orient the thrust vector in the required inertial direction. When the proper attitude is achieved, and at the correct time, midcourse burn is initiated. Attitude control during burn is again dependent on the inertial configuration chosen.

d) Subsequent cruise and midcourse correction phases: After completion of the first midcourse correction, the spacecraft will be "unwound" to the original Sun/Canopus reference attitude and continue in a cruise phase identical to the first. One or more further midcourse corrections will be made in a manner similar to the first. After the last midcourse correction, the trajectory will have been corrected such that terminal approach conditions meet the requirements.

e) Approach phase: On the premise that a mission with terminal approach requirements is more stringent than those imposed on Mariner 1969 or Voyager, it is postulated that some form of approach navigation will be required that is more accurate than that available with DSIF tracking. To cover this possibility, this study includes considerations of an approach sensor. The sensor to be considered will be a planet tracker with 2 deg of electronic scan freedom relative to the kick stage. The sensor can provide:

1) Stadimetric ranging data

2) Clock and cone angles relative to the Sun/Canopus frame of reference.

On the basis of approach measurements provided either via DSIF alone or in combination with the approach sensor, the approach trajectory will be determined and the following will be computed:

1) Magnitude and inertial direction of the deboost velocity to achieve the desired orbit about Mars

2) Time of initiation of the deboost thrust

Orbit determination accuracy requirements for the approach phase of the Voyager mission are discussed in par. 3.4.3. The accuracies achievable with and without an on-board planet tracker are analyzed in vol. II, subsec. 4.6.

f) Deboost velocity application: The sequence of events and operations of this phase is the same as for the midcourse correction phase. The deboost velocity increment is approximately 2.5 km/sec.

g) Orbiting phase: During the orbiting phase the spacecraft is attitude stabilized using the Sun and Canopus as attitude references. Tracking by DSIF is continued to refine the knowledge of the orbit parameters, and the results are used to calculate the orbital transfer maneuvers and the orbit trim manuevers required for achieving the final orbit. The accuracy of the orbit determination achievable for this mission is discussed in subsec. 5.8.

h) Orbit transfer and orbit trim maneuvers: Transfer from the initial 1100 x 10,000 km (periares and apoares altitudes) orbit to a lower orbit (e.g., 500 km altitude circular) is made via a two-burn transfer. The first burn lowers the periares altitude. The second burn circularizes the orbit at the desired altitude. The sequence of events for these maneuvers is the same as for midcourse corrections.

3.4.3 Guidance System Performance Requirements

3.4.3.1 Terminal Accuracy Requirements

Because of 1) midcourse correction execution errors, 2) imperfect approach trajectory estimation, and 3) execution errors at deboost burn, the final orbit will differ from the desired orbit. The Voyager mission requirements, translated into system accuracy requirements, for these three types of errors are summarized in the following paragraphs.

3.4.3.2 Accuracy Requirements for Interplanetary Trajectory Corrections, Mars Orbit Insertion, and Mars Orbit Trim

The required precision with which the spacecraft must execute the arrival date separation maneuver* and interplanetary trajectory

*The maneuver is intended to separate the arrival time of the two spacecraft launched by the Saturn V booster by at least 8 days. The maneuver is made in a similar manner, but prior to the first midcourse correction.

corrections is a complicated function involving a number of considerations such as the final mission accuracy requirements, the orbit-determination uncertainty as a function of time, the number of maneuvers to be performed, the amount of trajectory biasing necessary to satisfy the probability-of-impact constraint, the orbit trim philosophy, and several other considerations, all of which are vitally interwoven in the mission formulation. In order to achieve a rational balance among the various mission accuracy requirements, Table 3-V from Ref. 3-8 specified the maximum allowable maneuver errors during different phases of the flight. The maximum allowable maneuver errors are defined by an error ellipsoid with an axis of symmetry parallel to the specified velocity increment.

3.4.3.3 Orbit Determination Accuracy Requirements for Interplanetary and Mars Orbiting Phases

To meet the overall mission accuracy requirements given above, specification must also be placed on the orbit determination accuracy using the earth-based DSIF. Due to the trajectory geometry, tracking system characteristics, and the presence of trajectory disturbances, the orbit determination uncertainties vary throughout the mission. The allowable uncertainties (from Ref. 3-8) are shown in Tables 3-VI and 3-VII.

3.5 LUNAR ORBITER MISSION

The booster/payload combination for this mission is assumed to be the Atlas SLV3X/Centaur/HEUS/lunar orbiter. Based on past experience, e.g, Surveyor and Lunar Orbiter, the need for a sophisticated inertial guidance system on the spacecraft is questionable for the translunar and lunar operations phases of this mission. However, for this study, it is assumed that the kick stage guidance system is to be used not only for these phases but also for primary guidance and control of the lower booster stages. The operational sequences and functional requirements are summarized below.

3.5.1 Mission Characteristics

This mission is operationally very similar to the Mars Orbiter mission discussed in subsec. 3.4. The primary difference to be noted is that USBS/DSIF tracking and orbit estimation accuracy will probably be sufficient to obviate the need for an approach sensor.

TABLE 3-V

MAXIMUM VELOCITY ERROR ELLIPSOIDS (FROM REF. 3-8)

Maneuver	Minimum Velocity Increments† (m/sec)	Maximum Velocity Increments (m/sec)	3σ error component parallel to specified velocity increment (m/sec or % of specified velocity increment)	3σ error component normal to specified velocity increment along any two orthogonal axes (m/sec or % of specified velocity increment)
Interplanetary trajectory corrections	1.0 (0.3)	100 (200)	larger of 0.1 m/sec or 3.0% (larger of 0.03 m/sec or 2.0%)	larger of 0.1 m/sec or 3.0% (larger of 0.03 m/sec or 2.0%)
Mars orbit insertion	1.0 km/sec	2.0 km/sec	3.0% (1.5%)	5.0% (3.0%)
Mars orbit trim	5.0 (1.5)	150	larger of 0.5 m/sec or 5.0% (larger of 0.2 m/sec or 3.0%)	larger of 0.5 m/sec or 5.0% (larger of 0.2 m/sec or 3.0%)

Note: Numbers not in parentheses are maximum values.

Numbers in parentheses are design goals.

†For purpose of error calculations only.

TABLE 3-VI

MAXIMUM ALLOWABLE 3σ ORBIT DETERMINATION UNCERTAINTIES (INTERPLANETARY ORBIT INJECTION THROUGH MARS ENCOUNTER) (FROM REF. 3-8)

Time at which orbit estimate is calculated	Magnitude and time of prior orbit corruptions†	Uncertainty in magnitude of impact parameter vector (km)	Uncertainty in aiming plane normal to nominal impact parameter vector (km)	Uncertainty in time of encounter (min)
Injection +2 days	Planetary vehicle injection	2000 (1000)	2000 (1000)	15 (7)
Injection +30 days	150 m/sec arrival date adjustment and interplanetary trajectory correction at I +5 days	1000 (500)	1500 (750)	10 (5)
Encounter -30 days	5 m/sec interplanetary trajectory correction at I +30 days	500 (300)	750 (500)	4 (3)
Encounter - 2 days	1 m/sec interplanetary trajectory correction at E -30 days	400 (150)	500 (200)	3 (1)
Encounter - 4 hr	Same as above	300 (100)	500 (150)	2 (0.5)

†For purposes of error analysis only. Numbers not in parentheses are maximum allowable uncertainties. Numbers in parentheses are design goals.

TABLE 3-VII

MAXIMUM ALLOWABLE 3σ ORBIT DETERMINATION UNCERTAINTIES (MARS ORBITING PHASE) (FROM REF. 3-8)

Time at which orbit estimate is calculated	Magnitude and time of prior orbit corruptions†	Uncertainty in orbit semi-major axis (km)	Uncertainty in orbit eccentricity	Uncertainty in time of periapsis passage (sec)
Orbit insertion +4 returns to orbit periapsis	2.2 km/sec orbit insertion maneuver	10 (1)	10^{-4} (10^{-5})	5 (0.1)
Orbit trim +3 returns to orbit periapsis	Orbit trim maneuver of 100 m/sec	10 (1)	10^{-4} (10^{-5})	5 (0.1)

†For purposes of error analysis only. Numbers not in parentheses are maximum allowable uncertainties. Numbers in parentheses are design goals.

A parking orbit ascent trajectory with a coast time of approximately 14 min was selected for this study. The rationale for this selection was based on the fact that the largest figure-of-merit is obtained for parking orbit missions having coast times in this range.

The lunar mission reference trajectory used for error analysis purposes was a closed loop targeted trajectory for the Atlas Centaur (AC-12 Configuration) launch vehicle. The trajectory profile is shaped by a predetermined pitch steering program from launch to booster cutoff (BECO). After BECO the sustainer is ignited and closed loop guidance is initiated. The guidance system continues to steer the vehicle through sustainer cutoff (SECO) and Centaur first-burn ignition until parking orbit is reached. The first-burn duration (launch to parking orbit injection) is approximately 585 sec and injects the vehicle into a 167 km perigee, 173 km apogee orbit. The Centaur stage coasts in this orbit for 845 sec, whereupon it reignites and burns for another 106 sec, injecting the payload into a highly elliptical ($e = 0.97167$) transfer orbit. The transfer time is approximately 65 hr.

Two midcourse corrections are assumed for this mission, the first at 15 to 20 hr after injection, and the second a few hours prior to lunar intercept.

Deboost is made into an intermediate orbit with approximate apsis distances of 3590 and 1990 km. The deboost velocity increment required is 745 m/sec. After accurate determination of the orbit has been made, a final orbit adjust maneuver is made to place the vehicle into a 3589 by 1784 km orbit.

3.5.2 Guidance System Operational Sequence

The guidance system operational sequence for the various phases of the lunar orbiter mission is described below:

a) Launch and boost to ~ 167 km parking orbit: The kick stage strapdown inertial guidance subsystem will provide the guidance function for this phase.

b) Coast in parking orbit: The kick stage and payload will coast in the parking orbit until translunar injection, which occurs approximately 14 min after entering the parking orbit. The inertial guidance subsystem will be relegated to the role of an attitude reference during this phase.

c) Translunar injection: The kick stage will be ignited to inject the kick stage/payload into the translunar trajectory. Attitude and burn control will be provided by the strapdown inertial guidance subsystem.

d) Coast until first midcourse correction: Following the injection burn, a celestial reference acquisition sequence is initiated and the kick stage/payload will be attitude fixed to the sun and the star Canopus via body-fixed sun and star sensors. The strapdown accelerometers can be turned off (except for heaters), and the flight computer algorithm for updating the direction cosines can be placed in a standby mode.

Deep-Space Network (DSIF) tracking will be used during this coast phase for orbit determination and to compute the midcourse velocity correction required to reduce the effects of injection errors. The midcourse thrust vector pointing and magnitude commands and time of execution command will be transmitted to the kick stage system.

e) First midcourse correction: Approximately 15 to 20 hr from translunar injection, the first midcourse correction will be executed. Ten to 30 min prior to the time of execution, the accelerometers will be turned on, the direction cosine solution algorithm will be initialized, and the vehicle rotations will be commanded to orient the thrust vector in the required inertial direction. When the proper attitude is achieved, and at the correct time, the midcourse burn is initiated.

f) Second coast phase and second midcourse correction: After completion of the first midcourse correction, the kick stage/payload will be "unwound" to the original Sun/Canopus reference attitude and continue in a cruise phase identical to the first. The second midcourse burn will occur a few hours prior to lunar injection and is designed to null selected miss components at lunar intercept.

g) Coast until deboost maneuver into intermediate lunar orbit: This phase will be identical to the other coast phases.

h) Deboost into intermediate lunar orbit: Based on the tracking data obtained, the kick stage/payload will be injected into an intermediate orbit with approximate apsis distances of 3590 and 1990 km. The deboost velocity increment required is approximately 745 m/sec.

i) Coast in intermediate orbit: The amount of coast time in the intermediate orbit will be chosen such that the orbit is properly phased with respect to the preselected target. The kick stage/payload will be tracked by DSIF stations to determine orbital parameters and the retro-maneuver required to place the kick stage/payload into the final orbit.

j) Retro into final orbit: Based upon the orbital estimates obtained from DSIF tracking data, and controlled by the strapdown inertial guidance system, the spacecraft will be injected into the final orbit. The desired final orbit will nominally have an apocynthion and pericynthion of 3589 km and 1784 km, respectively.

3.5.3 Guidance System Performance Requirements

3.5.3.1 Translunar Injection

The kick stage/payload must be injected into a translunar trajectory such that the desired lunar orbit can be achieved by the kick stage propulsion capability. A set of deviations* of the kick stage/payload position and velocity from the nominal trajectory which will permit the meeting of the requirements of the final orbit is listed in Table 3-VIII.

3.5.3.2 Translunar Coast Phases

Prior to the first (second) midcourse corrections, the deviations of position and velocity from the nominal trajectory must be within certain limits. These limits are determined by the correction capability of the midcourse correction system. A set of injection deviations from the nominal trajectory propagated to the point of the first midcourse correction which satsify the midcourse correction capability are listed in Table 3-VIII. Prior to the second midcourse maneuver, the deviations must be such that the correction of miss components at the target are within the capability of the second midcourse maneuver. A set which satisfies these requirements is shown in Table 3-VIII.

*The position and velocity errors are stated either in an earth-centered inertial (ECI) coordinate system (X-axis in the direction of the vernal equinox) or in selenographic coordinates. Note that these errors are stated as deviations from the *a priori* nominal trajectory. See par. 1.4.5 for definition of coordinate systems.

During each phase, the position and velocity of the kick stage/payload will be determined by earth-based tracking stations. At the end of the final coast phase, as a result of the midcourse corrections, the position and velocity of the kick stage/payload must be within the limits shown in the last column of Table 3-VIII.

3.5.3.3 Midcourse Correction Maneuvers

Approximately 15 hr after translunar injection, the first midcourse correction will be commanded. The requirements on the maneuver execution errors are shown in Table 3-IX. The guidance law assumed is directed to nulling the errors in the impact plane and error in the time of flight or the impact plane error only.* Hence, these controlled quantities will be reduced by the midcourse maneuver.

The second midcourse will be executed a few hours prior to translunar injection. The requirements on the maneuver execution errors are in Table 3-IX.

3.5.3.4 Deboost into Lunar Orbit

Based upon tracking data, the following quantities will be determined for injection into the intermediate orbit:

a) Thrust initiation time

b) Body attitude

c) Velocity increment

These quantities will be computed to null the deviations from nominal of the apocynthion, inclination, longitude of the ascending mode, and the argument of pericynthion. A set of required accuracies of position and velocity at the end of this phase which meet the orbital requirements is given in Table 3-X.

*The performance analysis results are presented in sec. 4 of Ref. 3-1 for both guidance laws. Detailed mission payload requirements dictate the choice for a given mission.

TABLE 3-VIII

GUIDANCE REQUIREMENTS FOR TRANSLUNAR INJECTION AND TRANSLUNAR COAST PHASES (1σ VALUES)

Parameter	Translunar Injection	Coast Until First Midcourse Correction	Coast Until Second Midcourse Correction	Coast Until Deboost into Intermediate Lunar Orbit
ΔR	6.8 km	693.9 km	141.9 km	10.0 km
ΔT	24.5 km	1171.9 km	441.4 km	94.3 km
ΔN	17.3 km	277.7 km	90.7 km	7.9 km
$\Delta \dot{R}$	20.3 m/sec	19.5 m/sec	1.5 m/sec	35.5 m/sec
$\Delta \dot{T}$	9.2 m/sec	14.7 m/sec	5.5 m/sec	2.8 m/sec
$\Delta \dot{N}$	37.3 m/sec	4.2 m/sec	1.6 m/sec	8.8 m/sec
Coordinate System	RTN*	RTN	RTN	RTN

*See Paragraph 1.4.5 for definition of this coordinate system

TABLE 3-IX

GUIDANCE AND CONTROL REQUIREMENTS FOR FIRST AND SECOND MIDCOURSE CORRECTIONS

Parameter	First Midcourse Correction	Second Midcourse Correction
Pointing error	0.4 deg (1σ)	0.4 deg (1σ)
Error proportional to ΔV	0.04% (1σ)	0.04% (1σ)
Velocity cutoff Resolution error	0.02 m/sec (1σ)	0.02 m/sec (1σ)
Velocity increment required (not to be exceeded more than 1% of the time)	64 m/sec	3 m/sec

TABLE 3-X

GUIDANCE AND CONTROL REQUIREMENTS FOR DEBOOST INTO INTERMEDIATE AND FINAL LUNAR ORBIT

Parameter	Deboost Into Intermediate Orbit (1σ values)*	Deboost Into Final Orbit (1σ values)*
ΔR	25.4 km	14.5 km
ΔT	80.5 km	21.0 km
ΔN	0.9 km	8.9 km
$\Delta \dot{R}$	48.3 m/sec	13.1 m/sec
$\Delta \dot{T}$	2.6 m/sec	3.2 m/sec
$\Delta \dot{N}$	3.7 m/sec	0.1 m/sec
Pointing error	0.4°	0.4°
Error proportional to ΔV	0.04%	0.04%
Velocity cutoff	0.02 m/sec	0.02 m/sec
Velocity increment required (not be exceeded more than 1% of the time)	758 m/sec (2485 ft/sec)	33 m/sec (110 ft/sec)
Coordinate system	RTN	RTN

3.5.3.5 Coast in Intermediate Orbit and Final Orbit Insertion

There are no active guidance requirements during the intermediate orbiting phase. However, the position and velocity must be within certain limits at the end of this phase. A set of position and velocity accuracies which (in combination with the expected execution errors) will not violate the orbit accuracies required is indicated in Table 3-XI. Orbit determination accuracies achievable are discussed in subsec. 4.7 vol. II.

The required maneuver for final adjustment of the orbit will be calculated using previous estimates of position and velocity. The maneuver (pitch attitude, yaw attitude, velocity magnitude) will be calculated so as to null the deviations at the pericynthion after retrothrusting at apocynthion to give a specified pericynthion inclination and argument of pericynthion.

*All values are 1σ except the required velocity increment.

TABLE 3-XI

GUIDANCE REQUIREMENTS FOR COAST IN INTERMEDIATE ORBIT

Parameter	Specification (1σ Values)
ΔR	14.5 km
ΔT	21.1 km
ΔN	8.9 km
$\Delta \dot{R}$	5.8 m/sec
$\Delta \dot{T}$	3.8 m/sec
$\Delta \dot{N}$	0.8 m/sec
Coordinate System	RTN

At the completion of the maneuver, the position and velocity must be within prescribed limits so that the desired lunar orbit can be achieved. The final lunar orbital requirements are given in Table 3-XII.

TABLE 3-XII

LUNAR ORBITAL PHASE ACCURACY REQUIREMENTS

Parameter	Specification (1σ Values)
Error in semimajor axis	7.24 km
Error in pericynthion altitude	0.2 km
Inclination error	0.01 deg
Error in ascending node at first target pass	
• Selenographic latitude	0.1 deg
• Selenographic longitude	0.1 deg
Error in augument of periapsis at first target pass	0.01 deg

3.6 SOLAR PROBE WITH JUPITER ASSIST

3.6.1 Mission Characteristics

It has been shown (Ref. 3-9) that the gravitational field of Jupiter may be employed to obtain solar probe and out-of-ecliptic postencounter trajectories following a close flyby past that planet. A 1972 solar impact mission and a 1972 90° out-of-ecliptic mission have been analyzed assuming the Saturn IB/Centaur launch vehicle characteristics given in Table 3-XIII (Refs. 3-3, 3-10, and 3-11). Specific launch sequence event times are summarized in Table 2-VII, vol. II.

Following the Centaur second cutoff, the payload coasts in the heliocentric transfer ellipse to Jupiter encounter (see Figure 3-5). The earth-centered and heliocentric transfer trajectory characteristics of both missions are essentially the same. The altitude of closest approach at Jupiter and the components $\bar{B} \cdot \bar{T}$ and $\bar{B} \cdot \bar{R}$ of the impact parameter $\bar{B}$ determine the postencounter trajectories. The impact parameter, $\bar{B}$,

TABLE 3-XIII

SATURN IB/CENTAUR LAUNCH AND INJECTION TRAJECTORY CHARACTERISTICS

Phase	Duration (min)	Angle (deg)	Altitude (n. mi.)
Total powered flight	18.12	53.70*	
Circular parking orbit	1.08	4.42*	100.0
Injection		12.7**	313.0

*Angle traversed, measured in earth-centered inertial coordinates.

**Flight path angle at injection, measured (+) above the local horizontal.

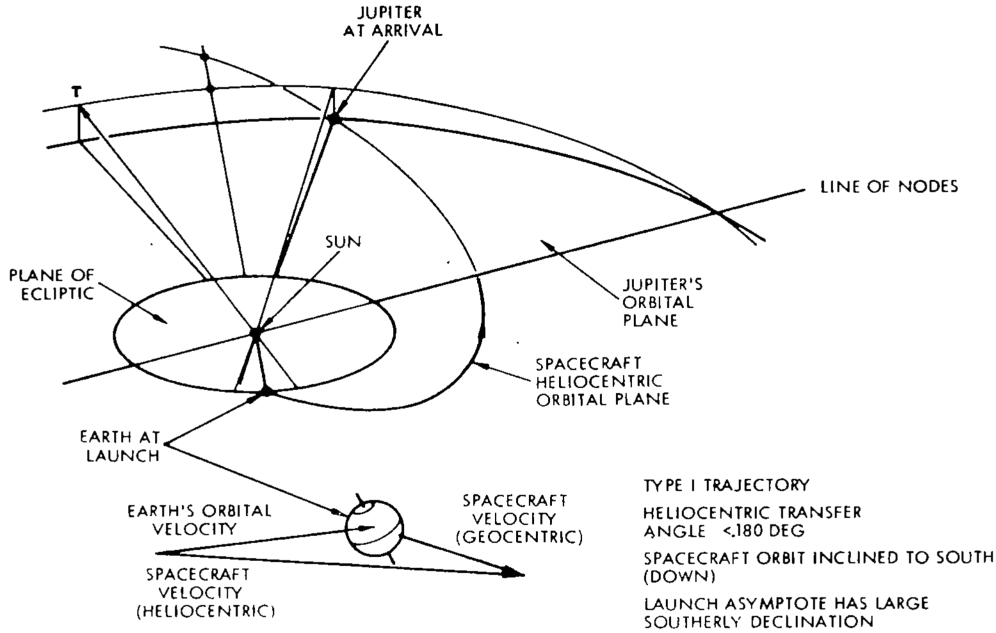

Figure 3-5. Typical Type I Earth-Jupiter Trajectory

is defined as a vector originating at the center of the target and is perpendicular to the incoming asymptote, $\bar{V}_\infty$ (see Figure 3-6). A unit vector $\bar{T}$ is defined as lying to a plane parallel to the ecliptic according to

$$T = \frac{\bar{V}_\infty \times \bar{k}}{|\bar{V}_\infty \times \bar{k}|}$$

where $\bar{k}$ is a unit vector normal to the ecliptic plane and pointing toward the north. The $\bar{R}$ axis is defined by

$$\bar{R} = \frac{\bar{V}_\infty \times \bar{T}}{|\bar{V} \times \bar{T}|}$$

The impact parameter, $\bar{B}$, lies in the $\bar{R}$-$\bar{T}$ plane and has components $\bar{B} \cdot \bar{T}$ and $\bar{B} \cdot \bar{R}$.

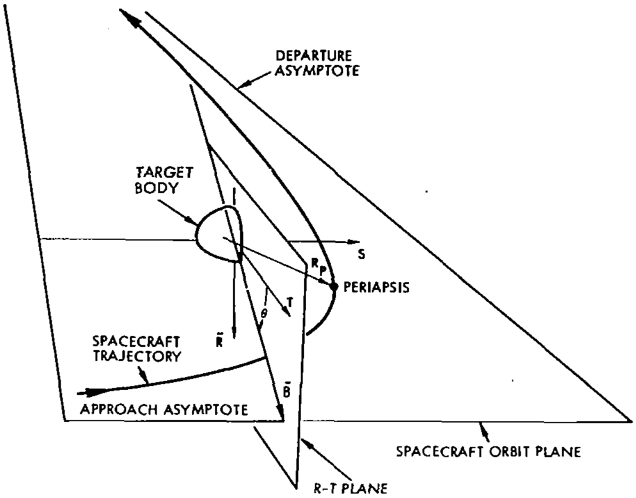

Figure 3-6. Encounter Geometry

Table 3-XIV summarizes the pertinent trajectory characteristics of each mission. The velocity of the solar probe as it becomes enveloped in the sun's photosphere is 617.45 km/sec; the total flight time beginning from injection is 2.762 yr.

3.6.2 Guidance System Operational Sequence

Independent of the specific mission trajectory chosen, the various mission profiles do not differ significantly from one another in the mission phases and required guidance system functions. The typical trajectory will contain the following phases with the indicated guidance system functions required:

a) Launch and boost to ~ 185 km parking orbit: The kick stage strapdown inertial guidance system will provide the guidance function for this phase.

b) Coast in parking orbit: Following injection into the parking orbit, the kick stage/payload will coast until the interplanetary orbit injection maneuver. The inertial guidance system will perform attitude reference and control functions during this phase.

TABLE 3-XIV

CHARACTERISTICS OF 1972 JUPITER PROBES

Departure date	1972 March 16	
Arrival date	1973 June 23	
Time of flight, days	463.97	
Departure asymptote (from earth)		
V_∞, km/sec	10.93	
C_3, km^2/sec^2	119.38	
Angle to equatorial plane, deg	-24.49	
Angle to sun-earth line, deg	254.54	
Heliocentric orbit		
True anomaly at departure, deg	4.799	
True anomaly at arrival, deg	57.593	
Heliocentric transfer angle, deg	128.21	
Inclination to ecliptic, deg	0.664	
Perihelion distance from sun, AU	0.987	
Aphelion distance from sun, AU	12.603	
Eccentricity	0.8547	
Approach asymptote (to Jupiter)		
V_∞, km/sec	13.99	
Angle to plane of Jupiter's orbit, deg	0.90	
Angle to plane of Jupiter-Sun line, deg	157.49	
Target parameters (at Jupiter)	Solar Probe	Out-of-Ecliptic Probe
Altitude of closest approach, Jupiter radii	3.03	6.23
$\bar{B} \cdot \bar{T}$, km	-674,781	-899,392
$\bar{B} \cdot \bar{R}$, km	14,787	-352,550

c) Heliocentric orbit injection: For the injection energy assumed, a velocity increment of approximately 7.0 km/sec is needed. This will be divided between the Centaur second burn and the kick stage. The kick stage inertial guidance system will provide the attitude and burn control for both stages.

d) Coast in heliocentric transfer ellipse and midcourse correction: These phases are similar to the corresponding phases for the Mars and Lunar Orbiter missions. The mid-course correction will occur 5 to 20 days from injection.

e) Coast to Jupiter encounter: The strapdown inertial guidance system will perform only attitude control functions during this phase, with the primary attitude reference being obtained from the body-fixed sun and Canopus sensors.

3.6.3 Guidance System Performance Requirements

3.6.3.1 Overall Mission Accuracy Requirements

For both the solar probe mission with Jupiter swingby and Jupiter flyby mission to observe the planet requires that the vehicle pass the planet at a prescribed point defined by the impact vector $\bar{B}$. Another major mission requirement is the midcourse correction capability of the spacecraft. The tolerances shown in Table 3-XV are typical values and have been used as requirements in this study.

TABLE 3-XV

ASSUMED JUPITER MISSION PERFORMANCE REQUIREMENTS

Parameter	Value or Tolerance
Tolerance on impact parameter $\bar{B}$ (3σ). (Nominal aim points are given in Table 3-XIV.)	$\bar{B}\cdot\bar{R}$ 10,000 km $\bar{B}\cdot\bar{T}$ 10,000 km
Maximum allowable ΔV for midcourse corrections (not to be exceeded more than 1% of the time)	100 m/sec

3.6.3.2 Interplanetary Trajectory Injection

The ascent guidance phase will include the atmospheric and exo-atmospheric ascent, the injection into a parking orbit, and the final injection into the heliocentric elliptic transfer orbit. The accuracy of the injection conditions can be traded off with the midcourse correction requirements. The requirements shown in Table 3-XVI are based on typical midcourse correction capabilities.

TABLE 3-XVI

INJECTION GUIDANCE REQUIREMENTS FOR THE JUPITER MISSION

Parameter	Specification (1σ values)
Error in velocity magnitude at injection	9.5 m/sec
Total velocity error perpendicular to the velocity direction	34.7 m/sec

3.6.3.3 Midcourse Corrections

Midcourse corrections are required to remove the terminal errors resulting from injection inaccuracies. The number and timing of these corrections are functions of the correction philosophy, the tracking system accuracy, and the trajectory or spacecraft constraints on the maneuver. For the purpose of this study, a particular correction philosophy, trajectory, spacecraft configuration, and single midcourse correction are assumed (see subsec. 4.2). The midcourse correction removes either the time-of-flight error and terminal errors in two mutually perpendicular directions or terminal errors only.

The requirements for execution of the midcourse maneuver are presented in Table 3-XVII.

TABLE XVII

GUIDANCE REQUIREMENTS FOR MIDCOURSE CORRECTION

Parameter	Specification (1σ Values)
Proportional error	0.75%
Pointing error	2/3 deg
Velocity cutoff resolution error	0.0188 m/sec

Section 3

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3-5 "Burner II General Description," The Boeing Company, D2-82601-1, May 1966.

3-6 "Voyager Spacecraft Phase B, Task D, Final Report, Vol. II, Spacecraft Design and Performance," TRW Systems Report prepared for Marshall Space Flight Center, October 1967.

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3-8 "Performance and Design Requirements for the 1973 Voyager Mission, General Specification for," JPL, Document SE002BB001-1B21, (Draft), 1 January 1967.

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3-11 "Centaur Technical Handbook," Revision B, General Dynamics/Astronautics, GD/A-BPM64-001-1, 24 January 1966.

4. SUMMARY OF GUIDANCE AND CONTROL SYSTEM CONCEPTUAL DESIGNS

4.1 INTRODUCTION

This section presents the recommended guidance and control system conceptual designs for each of the five missions considered in this study. A conceptual design is defined as a functional representation of the component configuration responsive to a specific mission, and consists of the following:

1) A functional schematic of the complete guidance, navigation, and control system indicating all informational loops.

2) Performance descriptions of each component and component subsystem in terms of its functional description, accuracy, physical parameters, and reliability.

3) Statement of development status of each component.

Functional schematics for each of the missions are presented in subsecs. 4.2 through 4.6. Component descriptions and performance characteristics supporting the conceptual designs are summarized in sec. 7.

The guidance and control conceptual designs summarized in this section are based on the operational sequences and the guidance performance requirements developed under Tasks I and II (Ref. 3-1). These requirements have been refined and extended to reflect the revised mission definitions* and the five specific launch vehicle/payload combinations defined in subsec. 1-2.

The guidance system core concept adopted during the Tasks I and II studies was retained in this study. However, some of the basic functional concepts have been modified. In particular, the utilization of the inertial measurement unit and digital computer of the core configuration was extended to cover the launch and boost phases of all the missions.

*The characteristics of the missions that differ from those that were used in the Task I and II studies (Ref. 4-1) are described in sec. 2 of vol. II. A summary of the characteristics of all the missions is given in sec. 3 of this volume.

This modification of concept was made to examine the feasibility of performing launch/boost/injection guidance and control with a strapdown inertial package. In most instances it is difficult to justify (on the basis of cost, performance requirements, and payload weight and size) the need, or use, for a complete three-axis inertial measurement unit to be used solely for attitude control and midcourse velocity corrections in interplanetary missions. The addition of the launch and boost-phase guidance and control functions to the total set of functions to be performed by the system thus provides a tenable basis for including the three-axis inertial measurement unit for these missions.

The recommended conceptual guidance and control system configuration developed in this study for the boost vehicles considered herein ignores the basic fact that all these boosters already have highly developed or proven guidance packages of their own. However, it was not intended to propose replacement of the existing systems with the strapdown system of this study. Rather, the boosters used in this study served primarily as vehicles or bases from which the analytical and preliminary design studies could proceed.

With the above premise and based on the performance analyses the composite conceptual equipment configuration summarized in sec. 2 was developed. Discussions of each mission are presented in the following sections.

4.2 EARTH LOW-ALTITUDE POLAR-ORBIT MISSION

The powered and coast phases of the near-earth polar-orbit mission up to injection of the payload into the design orbit is of short duration (19.2 min) with no inordinate demands exceeding state-of-the-art guidance capabilities. Electro-optical sensors are not required for any mission phase; therefore, guidance system for this mission is comprised of only the core package.

The integrated guidance and control configuration is indicated in Figure 4-1. The basic guidance package is installed in the Burner II and provides the guidance function for the Atlas stages as well. A control electronics package is required on the Burner II to interface between

1) the primary ROI computer and the Burner II attitude control system and
2) the ROI computer and the Atlas components, indicated in Figure 4-1,

which are part of the existing Atlas system. The guidance performance analysis of this conceptual design can be found in subsec. 5.2.

Autopilot stability studies indicate that the Atlas rate gyros should be retained, with considerations of possible relocation (see sec. 6, vol. II). However, the Atlas position gyro functions can be taken over by the ROI core package. These comments pertaining to the Atlas hold for the two missions discussed in subsecs. 4.3 and 4.4.

4.3 EARTH-SYNCHRONOUS ORBIT MISSION

The integrated guidance and control conceptual configuration for the earth-synchronous orbit mission is indicated in Figure 4-2. An earth horizon scanner and a solar aspect sensor have been added to the core package. The core package, the electro-optical sensors, and an interface electronics package are installed on the Centaur. No changes are made to the basic Centaur control actuation system, and the Atlas control system configuration is the same as in the previous mission.

The functioning of the various sensors can best be described with reference to the basic mission profile. During the Atlas and first Centaur burns to parking orbit injection, guidance and steering are controlled inertially. For the direct-ascent mission, the second Centaur burn is initiated at first equatorial crossing, approximately half an hour after launch. During the intermediate coasting period, constant attitude is maintained and the second Centaur burn for Hohmann transfer from parking orbit altitude to synchronous altitude is again controlled inertially. For this direct-ascent mission, no external attitude or timing update information is required (Ref. 4-1).

However, for the long parking orbit coast case, both an attitude and timing update are highly beneficial prior to the second Centaur, or perigee, burn. Both these updates can be obtained with the combination of the earth sensor and solar aspect sensor shown in Figure 4-2.

During the long (approximately 5.25-hr) Hohmann transfer coast to apogee at synchronous altitude, attitude is maintained inertially. However, prior to the third Centaur, or apogee, burn, an attitude update is accomplished again with the aid of the earth and sun sensors. The performance

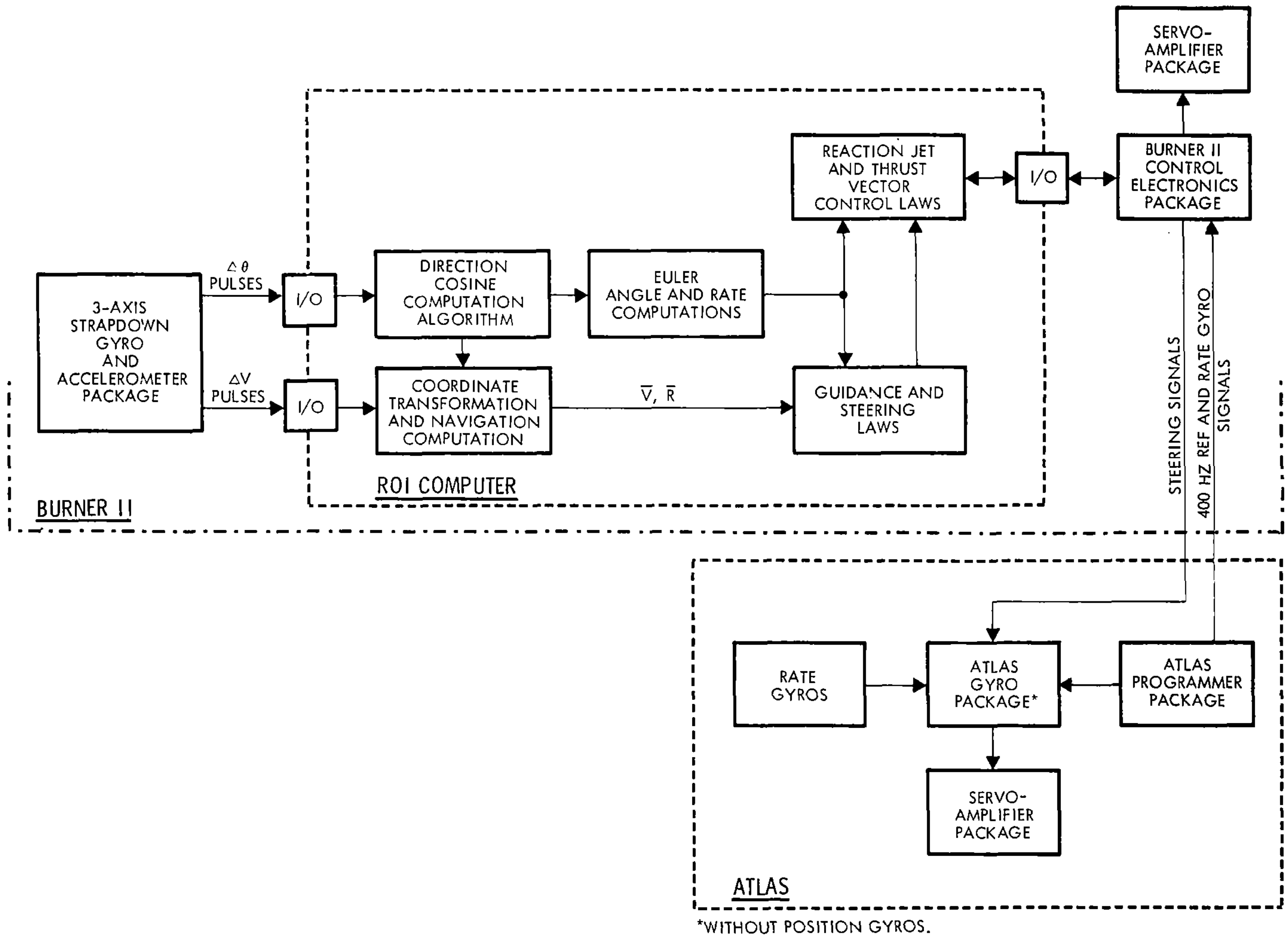

Figure 4-1. Basic Conceptual Design Configuration for the Near-Earth Polar-Orbit Mission

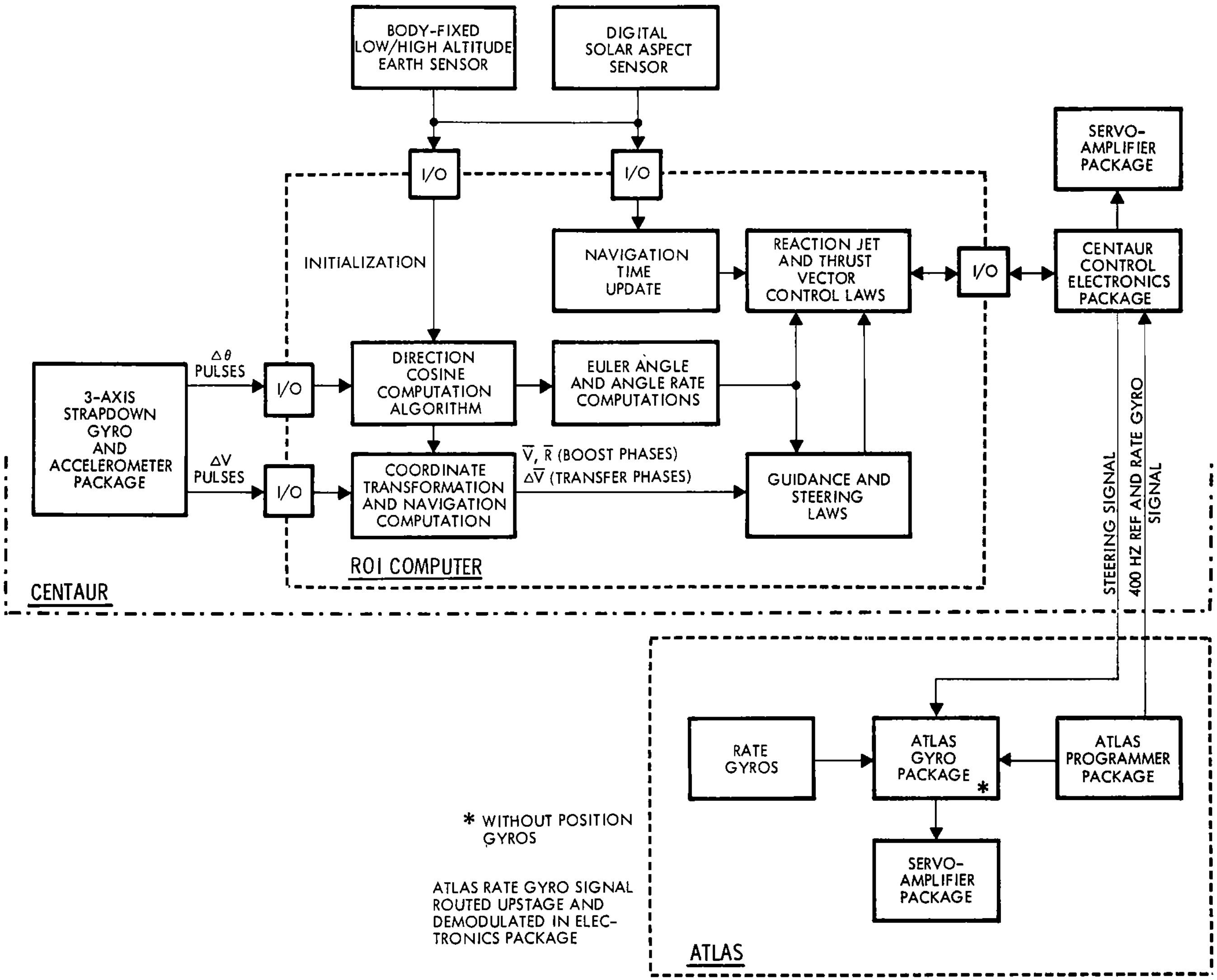

Figure 4-2. Basic Conceptual Design Configuration for the Earth-Synchronous Orbit Mission

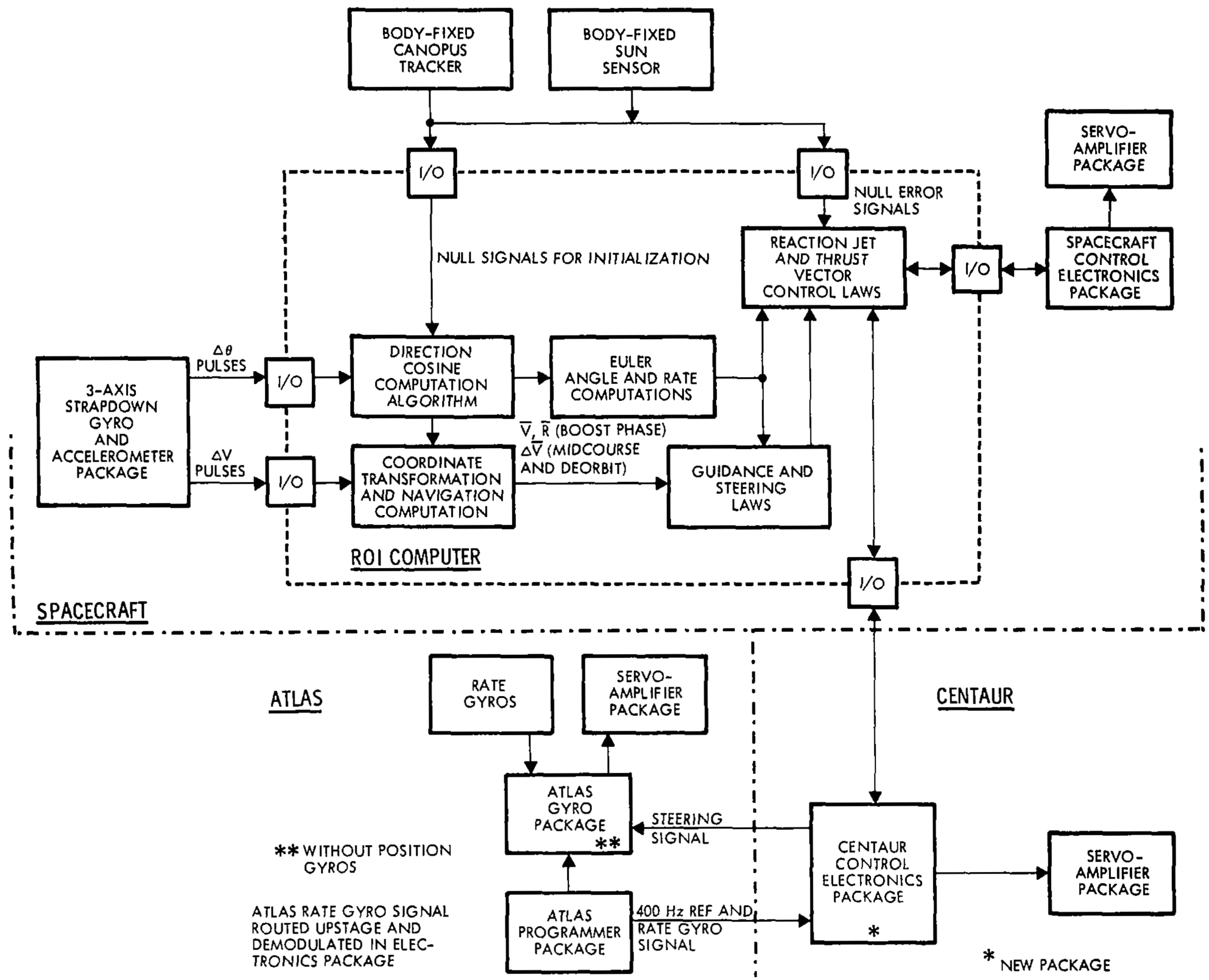

Figure 4-3. Basic Conceptual Design Configuration for the Lunar Orbiter Mission

achievable with this system configuration is summarized in subsec. 5.3.

4.4 LUNAR ORBITER MISSION

For the Lunar orbiter mission, the ROI guidance package is installed in the orbiter spacecraft, and data and signal transfer to the Atlas control system configuration is effected through a Centaur electronics interface package (Figure 4-3). The canopus tracker and sun sensor replace the earth sensor and solar aspect sensor of the previous mission. These sensors are used to establish the celestial attitude reference only during the translunar coasting phases.

Performance analyses for this mission were conducted during the Tasks I and II phases of this overall study. A summary of the translunar orbit injection analysis is presented in subsec. 5.4.

4.5 MARS ORBITER MISSION

The major difference in the system elements for the Mars orbiter mission as compared to those of the lunar orbiter mission is the possible addition of the planetary approach sensor. Data from this sensor, in conjunction with data from the sun and Canopus sensors, can be utilized by ground-based stations to improve the quality of the determination of the spacecraft approach orbit to Mars. However, for mission requirements comparable to those in use up to now, it is not clear that this improvement in approach orbit determination is absolutely essential. Thus, the planetary approach sensor shown in Figure 4-4 is included conditionally so that the implications on preliminary modular design can be investigated for applications to possible future missions with high-accuracy requirements.

Except for the planetary approach sensor, the functions and utilization of the total guidance and control system substantially paralleled the functional operations of the lunar orbiter mission. The Saturn V rate gyros are retained to simplify the autopilot design problem.

4.6 SOLAR PROBE (WITH JUPITER ASSIST) MISSIONS

Up to Jupiter encounter, the Jupiter flyby missions closely resemble the lunar mission. Therefore, the conceptual configuration, Figure 4-5, is very similar to that of the lunar mission.

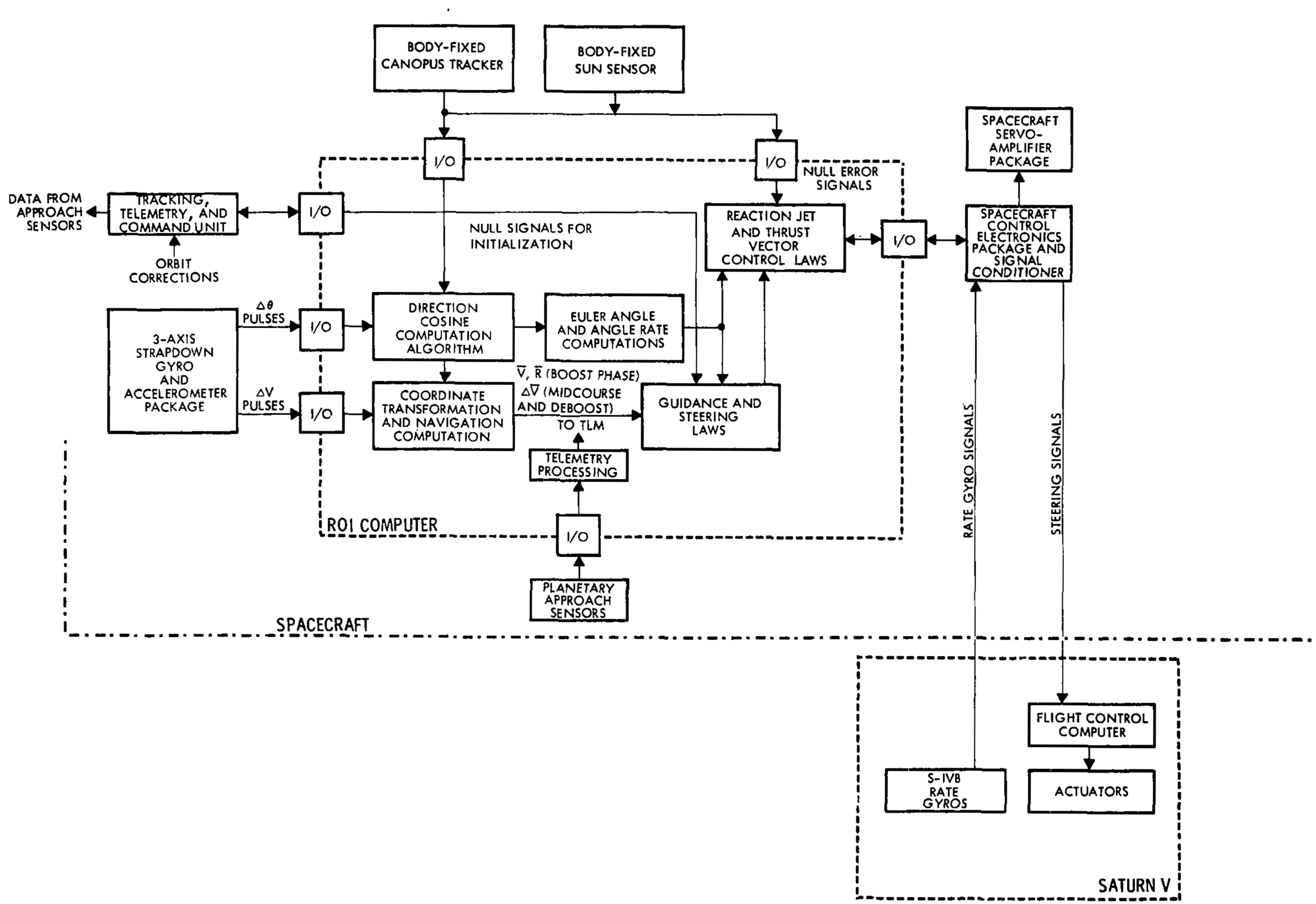

Figure 4-4. Basic Conceptual Design Configuration for the Mars Orbiter Mission

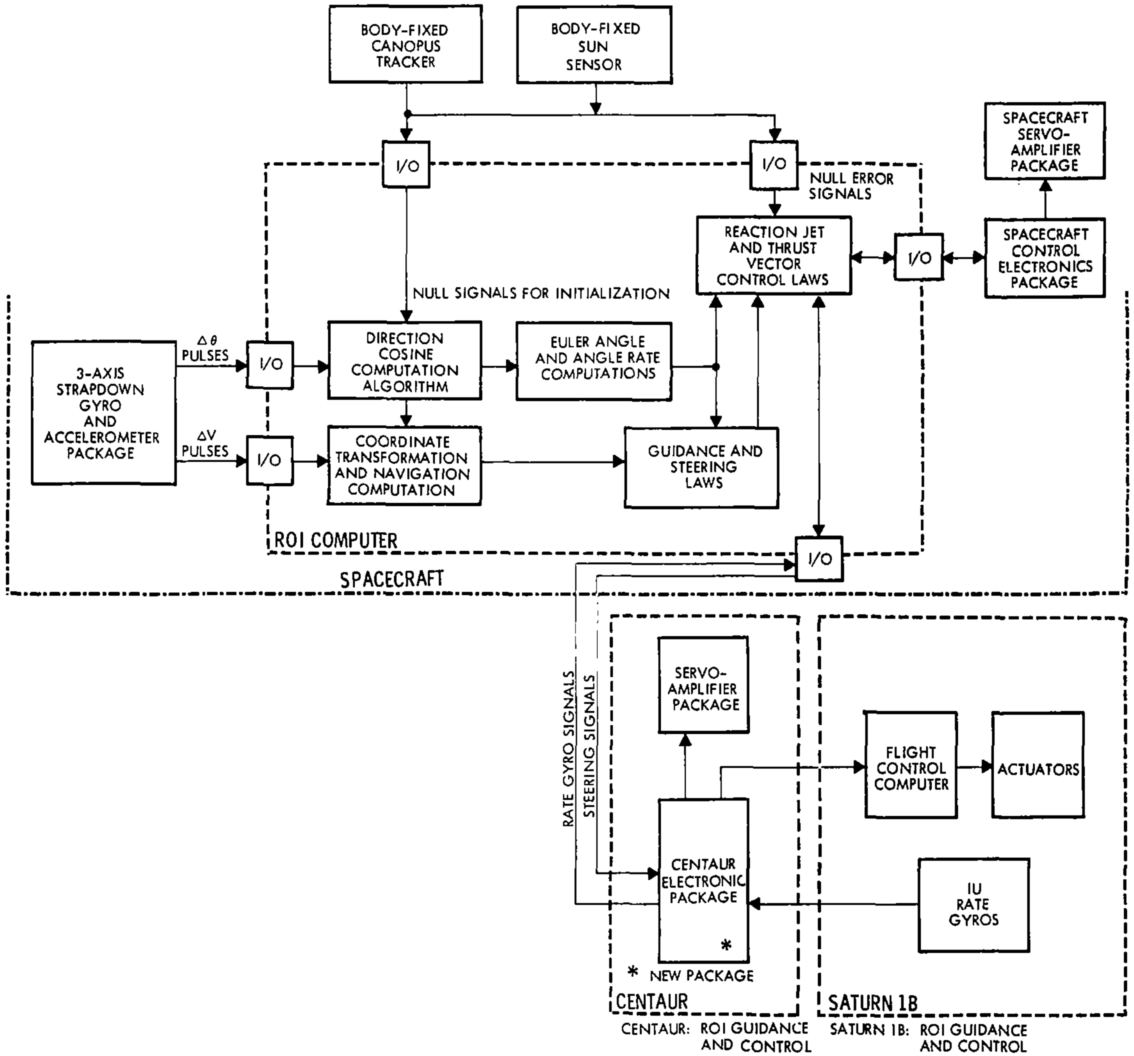

Figure 4-5. Basic Conceptual Design Configuration for the Jupiter Missions

REFERENCES

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5. SUMMARY OF GUIDANCE SYSTEM PERFORMANCE ANALYSES RESULTS

5.1 INTRODUCTION

This section summarizes the performance analyses of the candidate strapdown inertial guidance systems augmented as necessary by electro-optical sensors. The analyses include injection accuracies and corrective incremental velocity requirements for all missions, the midcourse and planetary insertion maneuvers for the lunar and interplanetary missions, and the Mars approach navigation analysis.

For each boost and injection analysis, a nominal trajectory (launch through injection) was generated which was representative of the mission desired, and an error analysis tape containing the position, acceleration, and attitude history for the powered-flight phase was produced for input to the error analysis program. The general characteristics of these powered-flight trajectories are given in sec. 2, Ref. 5-1, and sec. 3 of this volume. The navigational errors of the inertial guidance subsystems, as augmented by the optical sensor subsystem, were determined by means of an inertial guidance error analysis program which calculates the effect by integrating the first-order perturbation equations along a nominal trajectory. The error analysis computer program is described in par. 7.1.2., Ref. 5-1, and in Ref. 5-2.

The tracking and navigation error analyses for Mars approach were conducted using the SVEAD computer program. SVEAD is a state variable estimation and accuracy determination program (Ref. 5-3). The equations for the error analysis program are discussed in detail in Appendix D, Ref. 5-1. Further discussion is found in sec. 4, vol. II of this report.

The major inertial and optical error sources for the analyses are summarized in Table 5-I. More detailed error source breakdowns can be found in sec. 4, Ref. 21, and in secs. 4 and 5 of vol. II. For the boost and injection powered-performance analyses, the system initialization errors shown in Table 5-I were used. The initial orientation errors include, in addition to the values shown in the table, the effects of accelerometer errors. These effects are introduced because it is assumed that the accelerometers are used in a leveling mode to initialize the direction cosine matrix. The initialization accuracy of the direction cosine matrix in azimuth is varied parametrically.

TABLE 5-I

INITIAL CONDITION ERROR MODEL USED FOR STRAPDOWN INERTIAL GUIDANCE PERFORMANCE ANALYSIS

Description	Value
Velocity relative to earth	0
Vertical position	3.0 m
East, north position	15 m
Orientation (level)	20 arc sec
Orientation (azimuth)	Variable

5.2 POWERED FLIGHT PERFORMANCE ANALYSES OF THE NEAR-EARTH POLAR-ORBIT MISSION

For the near-earth polar-orbit mission, no optical sensors are required in the basic guidance system. An error analysis run was made for each of the two inertial system error models corresponding to the TG-166 and TG-266.

Summary of Results and Conclusions

For an initial azimuth alignment error of 20 arc sec, the resulting one-sigma rss position and velocity component errors in radial, tangential, normal (RTN) coordinates for the two inertial systems are as indicated in Table 5-II. (See par. 1.4.5 for definition of the RTN coordinate system).

TABLE 5-II

ATLAS/BURNER II NEAR-EARTH POLAR-ORBIT INJECTION ERRORS

System	Position (km)			Velocity (m/sec)		
	R	T	N	R	T	N
TG-166	1.28	1.59	4.01	3.28	1.50	6.05
TG-266	0.54	0.96	1.32	1.56	0.86	1.79

The major contributors to these totals are summarized in Tables 4-III and 4-IV, vol. II, for the TG-166 and TG-266 systems, respectively.

To relate these injection errors to mission performance, the 95% corrective ΔV required to correct the payload orbit was computed for both systems and for a range of initial azimuth accuracies. The results from a 1000-run Monte Carlo analysis are indicated in Figure 5-1. Both the average ΔV and 95% ΔV requirements are shown. As expected, the TG-266 system shows a performance effectiveness two to three times better than the TG-166. Also significant is the fact that the "knee" of the 95% ΔV curves occurs near 20 arc sec. Operationally, this value can be achieved by optical means and is a recommended prelaunch value.

5.3 ERROR ANALYSIS FOR SYNCHRONOUS ORBIT INSERTION

The synchronous orbit mission involves extended flight times so that a pure inertial system can cause unacceptable injection errors. Both optical attitude updates using onboard sensors and an autonomous time of perigee burn update are considered as solutions to the problem.

It is assumed that optical attitude update measurements may be made in the 185-km coasting orbit 10 min before perigee burn and in the Hohmann transfer orbit 10 min before apogee burn. The earth sensor errors are assumed to be 12 arc min/axis in 185-km orbit and 10 arc min/axis in synchronous orbit. The sun sensor errors are assumed to be 3 arc min/axis. It is assumed that the sun lies approximately in the direction of the vehicle-roll axis during the apogee measurement and fairly near the horizontal plane in the perigee measurement. The sun sensor is used for pitch and yaw angles in the apogee measurement and for the yaw angle in the perigee measurement, with the earth sensor being used for the remaining angles. The sun and earth sightings prior to perigee burn are used to determine sun zenith angle and autonomously predict the time of equatorial crossing. The accuracy with which this can be done is primarily a function of optical sensor errors. In subsec. 4.3, vol. II, the time update error is shown to be 4 sec.

The time updating is based on the geometry and formulation indicated in Figure 5-2. From accuracy considerations, the zenith angle A should

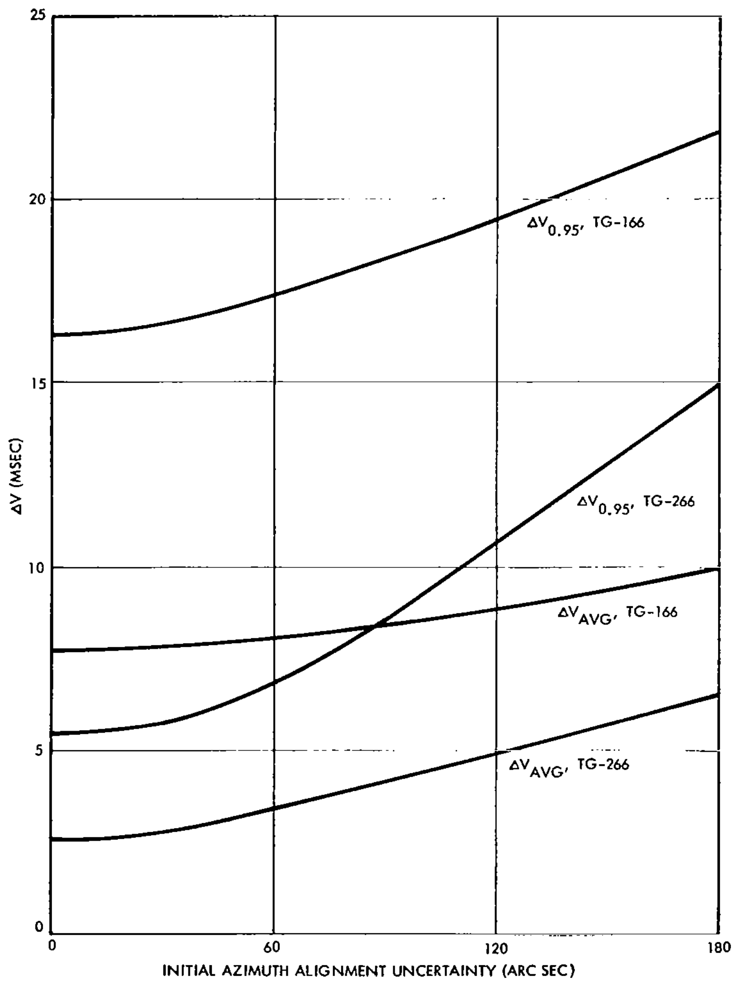
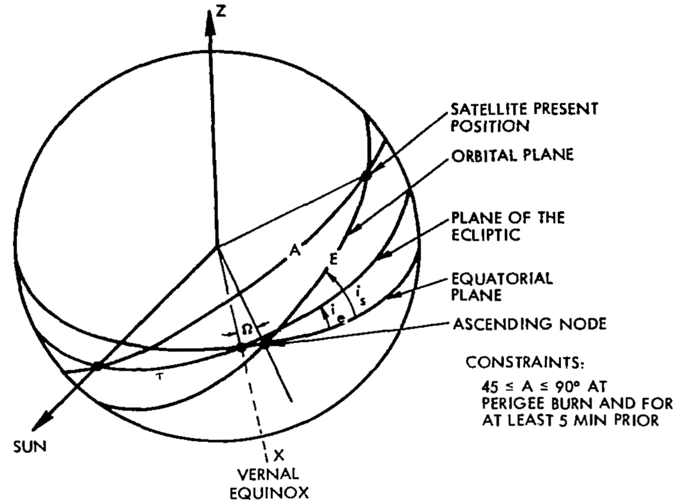


Figure 5-1. Corrective Velocity Requirements for Atlas/Burner II Near-Earth Polar-Orbit Mission



$$\cos A = (\cos \Omega \cos \tau - \sin \Omega \sin \tau \cos i_e) \cos E - \left[(\sin \Omega \cos \tau + \cos \Omega \sin \tau \cos i_e) \cos i_s + \sin \tau \sin i_e \sin i_s \right] \sin E$$

Figure 5-2. Sun Sighting Time Update Technique for Multi-Parking Orbit Synchronous Satellite Mission

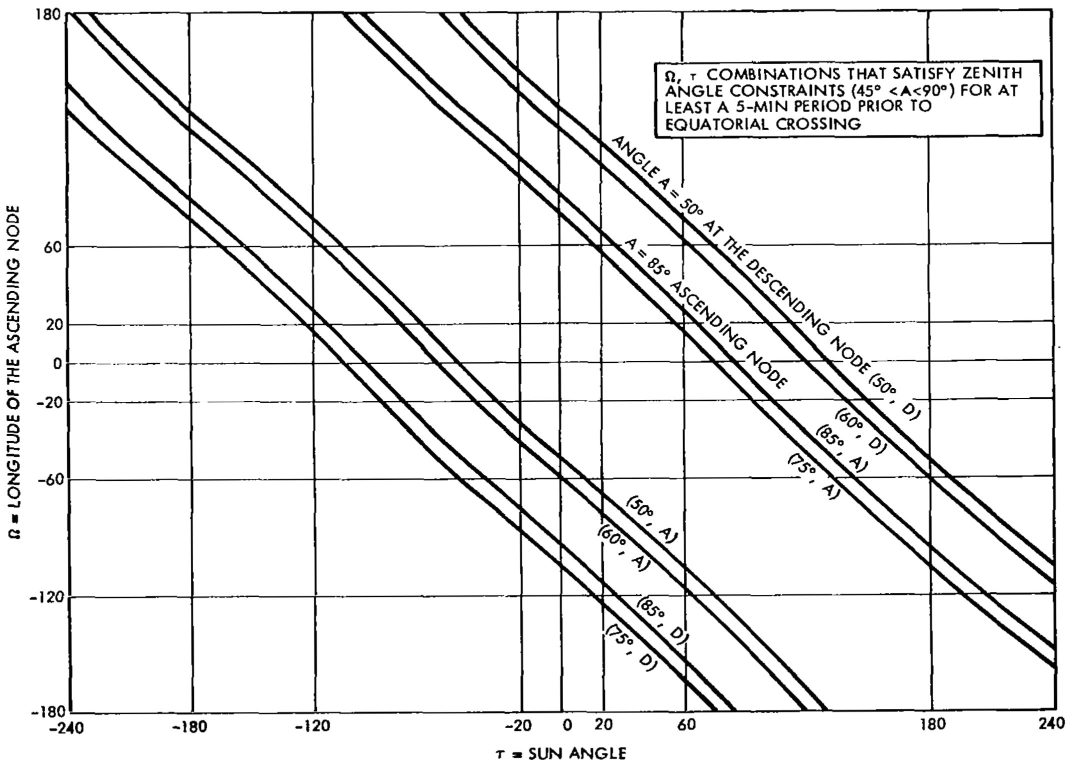


Figure 5-3. Combinations of Ω and τ that Satisfy the Visibility Constraints

be between 45° and 90°. The combination of sun angles and longitude of the ascending node which satisfy this constraint are illustrated in Figure 5-3.

Summary of Results and Conclusions

In the synchronous orbit mission, the errors at injection into synchronous orbit were first calculated. The delta-velocity required to achieve the desired orbit was then determined by Monte Carlo techniques. Twelve different runs were made with different candidate systems.

Table 5-III identifies the 12 runs made and Table 5-IV presents the results of these runs. One-sigma position, velocity, and orientation errors at injection into synchronous orbit are presented in RTN coordinates along with the ΔV required for 95% probability of successful synchronization.

An identification of the largest instrument error sources contributing to the position, velocity, and orientation errors is given in sec. 4, vol. II.

The following conclusions were reached for the synchronous orbit mission:

- Prelaunch calibration is desirable.
- Apogee attitude update is necessary for all missions.
- Perigee attitude update and time update are necessary for missions with a 185-km altitude coast period of long duration.
- The performance of the TG-166 system for long coasts, and of the more accurate TG-266 system for both short and long coasts, is limited by the horizon tracker errors.
- Time update errors of the magnitude used are not significant compared to other error sources.
- A full position and velocity update would not provide significant improvemenunless the attitude update errors were reduced.

5.4 ERROR ANALYSIS FOR TRANSLUNAR ORBIT INSERTION

The lunar mission was analyzed from liftoff to injection into the translunar orbit. The ΔV required for a 95% probability of successfully performing the midcourse correction is taken as a figure of merit.

TABLE 5-III

SYNCHRONOUS MISSION RUNS

Run No.	Coast Orbits	System No.	Prelaunch Calibration	Time Update	Attitude Update	
					Perigee	Apogee
1	0	166	No	No	No	No
2	0	166	No	No	No	Yes
3	0	166	Yes	No	No	No
4	0	166	Yes	No	No	Yes
5	0	266	No	No	No	No
6	0	266	No	No	No	Yes
7	8	166	No	No	Yes	Yes
8	8	166	No	Yes	Yes	Yes
9	8	166	Yes	No	Yes	Yes
10	8	166	Yes	Yes	Yes	Yes
11	8	266	No	No	Yes	Yes
12	8	266	No	Yes	Yes	Yes

TABLE 5-IV

ERROR ANALYSIS RESULTS FOR THE SYNCHRONOUS MISSION (RTN COORDINATES)

Run No.	Position (km)			Velocity (m/sec)			Orientation (arc sec)			95% ΔV (m/sec)
	R	T	N	R	T	N	Yaw	Roll	Pitch	
1	56.5	41.8	35.7	26.7	11.2	23.4	2900	3110	3670	73
2	56.7	41.8	35.7	7.4	1.9	1.9	176	505	308	13
3	49.7	20.8	19.8	26.4	11.0	23.6	2930	3090	3670	75
4	50.0	20.8	19.9	6.2	1.8	1.6	176	490	307	9
5	30.2	20.3	14.0	13.0	5.2	11.1	1380	1500	1760	35
6	30.3	20.3	14.1	5.2	1.1	1.2	136	482	285	8
7	513	793	430	83.5	14.5	8.1	176	505	308	163
8	59.5	148.2	84.3	10.4	2.0	2.1	176	505	308	23
9	354	534	290	56.7	10.2	5.8	176	490	307	109
10	53.2	136.2	78.2	9.7	1.9	2.1	176	490	307	20
11	259	408	222	42.7	7.4	4.4	136	482	285	83
12	33.6	142.2	77.6	8.7	1.3	1.8	136	482	285	20

The performance of the TG-166 and TG-266 systems was compared with that of a Centaur gimbaled inertial guidance system. Table 5-V presents the error model for the Centaur A/C-10 gimbaled IMU as obtained from Ref. 5-4. Figure 5-4 shows the Centaur gyro and accelerometer orientation at launch.

TABLE 5-V

ERROR MODEL FOR THE CENTAUR IMU
(from Ref. 5-4)

Type	Description	Value	Units
Initial	Vertical position	3.0	m
Initial	East, north position	15.3	m
Initial	Azimuth error	18.6	arc sec
Initial	Level errors	11.1	arc sec
Accelerometer	Bias	42	μg
Accelerometer	U accelerometer inflight bias	24	μg
Accelerometer	V accelerometer inflight bias	26	μg
Accelerometer	W accelerometer inflight bias	29	μg
Accelerometer	Scale factor	51	μg/g
Accelerometer	V accelerometer input axis rotation toward U axis	10.3	arc sec
Accelerometer	W accelerometer input axis rotation toward U axis	10.3	arc sec
Accelerometer	W accelerometer input axis rotation toward V axis	11.3	arc sec
Accelerometer	Scale factor g proportional nonlinearity	9.4	$\mu g/g^2$
Accelerometer	Output axis g^2 sensitivity	9	$\mu g/g^2$
Accelerometer	Input-pend. g-product sensitivity	13	$\mu g/g^2$
Accelerometer	Input-output g-product sensitivity	12	$\mu g/g^2$
Accelerometer	Pend. -output g-product sensitivity	8	$\mu g/g^2$
Gyro	U gyro bias drift	0.084	deg/hr
Gyro	W gyro bias drift	0.094	deg/hr
Gyro	V gyro bias drift	0.093	deg/hr
Gyro	U gyro input g-sensitive drift	0.106	deg/hr/g
Gyro	W gyro input g-sensitive drift	0.114	deg/hr/g
Gyro	V gyro input g-sensitive drift	0.101	deg/hr/g
Gyro	U gyro spin g-sensitive drfit	0.173	deg/hr/g
Gyro	W gyro spin g-sensitive drift	0.177	deg/hr/g
Gyro	V gyro spin g-sensitive drift	0.190	deg/hr/g
Gyro	Input-spin g-product drift	0.009	$deg/hr/g^2$

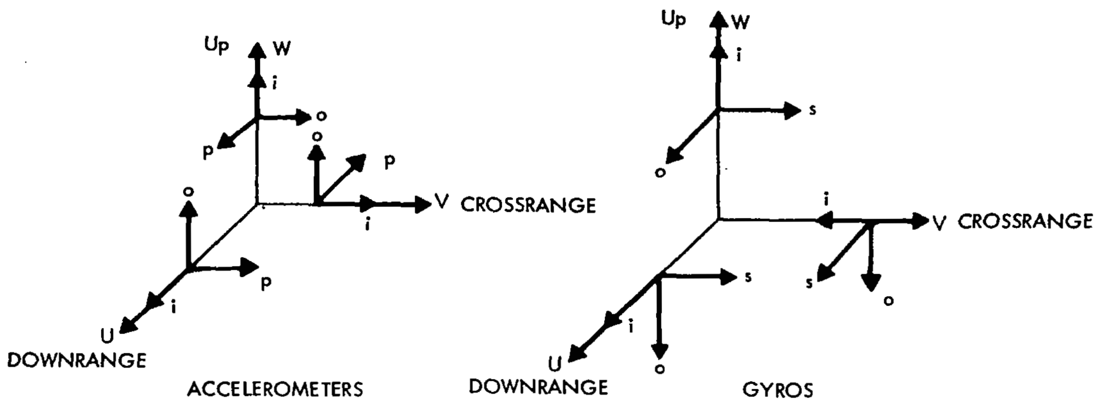


Figure 5-4. Centaur Sensor Orientation

Table 5-VI identifies the four runs made and summarizes the one-sigma position, velocity, and orientation errors at injection into earth-moon transfer orbit, and the ΔV required for 95% probability of successfully performing the midcourse correction. The errors are presented in both ECI (Earth Centered Inertial)* and RTN coordinates.* The ΔV requirement is given for the two cases of variable time of arrival guidance and fixed time of arrival guidance. Additional detailed results are presented in sec. 7, (Ref. 5-1).

The following conclusions were reached for the translunar orbit injection mission.

- Prelaunch calibration is desirable.
- The most significant error sources are pitch gyro bias and roll gyro mass unbalance for the strapdown systems and y-gyro mass unbalance for the gimbaled system.
- All resulting errors are well within the requirements summarized in Tables 3-V and 3-VI of sec. 3.

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*See par. 1.4.5 for coordinate system definitions. In this section the X-axis of the ECI coordinate system lies along the Greenwich meridian at launch.

TABLE 5-VIa

ERROR ANALYSIS RESULTS FOR THE LUNAR MISSION (RTN COORDINATES)

Run No.	System No.	Pre-launch Cal.	Position (km)			Velocity (m/sec)			Orientation (arc sec)			95% ΔV (m/sec)	
			R	T	N	R	T	N	Yaw	Roll	Pitch	Variable Time	Fixed Time
1	TG-166	No	1.8	3.5	5.6	7.5	2.9	9.0	568	233	304	50	62
2	TG-166	Yes	1.4	3.4	3.3	7.2	2.8	4.8	291	233	304	49	60
3	TG-266	No	0.8	1.9	1.8	3.6	1.4	2.8	175	114	147	25	31
4	A/C-10	No	1.5	3.4	1.5	5.0	2.4	2.4	161	152	150	37	48

TABLE 5-VIb

ERROR ANALYSIS RESULTS FOR THE LUNAR MISSION (ECI COORDINATES)

Run No.	System No.	Pre-launch Cal.	Position (km)			Velocity (m/sec)			Orientation (arc sec)		
			X	Y	Z	X	Y	Z	X	Y	Z
1	TG-166	No	2.7	3.9	5.0	8.1	4.8	7.7	550	253	319
2	TG-166	Yes	2.1	3.2	3.0	7.5	3.0	4.1	299	249	283
3	TG-266	No	1.2	1.8	1.7	3.8	1.7	2.3	172	123	143
4	A/C-10	No	2.0	2.9	1.8	5.2	2.1	2.1	160	152	152

5.5 ERROR ANALYSIS FOR INTERPLANETARY ORBIT INSERTION

The Mars and Jupiter missions were analyzed from liftoff to injection into the interplanetary orbit. Again the ΔV required for a 95% probability of successfully performing the midcourse is taken as a figure of merit. Both the TG-166 and TG-266 systems were evaluated but no comparison was made against existing Saturn V and Saturn IB/Centaur guidance systems. However, the 95% ΔV for both miss only (variable time) and miss plus time of arrival (fixed time) corrections were obtained. Initial azimuth alignment was varied parametrically for the Mars missions.

The one-sigma position and velocity errors at injection for the Mars mission are summarized in Table 5-VII. The uncorrected miss ellipses at Mars due to these errors are illustrated in Figures 5-5 and 5-6.

TABLE 5-VII

SATURN V MARS MISSION INJECTION ERRORS
(INITIAL AZIMUTH ALIGNMENT ERROR = 20 ARC SEC)

	Position (km)			Velocity (m/sec)		
	R	T	N	R	T	N
TG-166	7.1	2.01	3.37	7.96	2.89	4.84
TG-266	3.37	1.02	1.07	3.52	1.30	1.61

As the initial azimuth alignment value is varied parametrically, only the normal components of position and velocity in Table 5-VII vary. The variation is as indicated in Figure 5-7. The 20-arc sec value again appears to be a satisfactory compromise between operational feasibility and system performance. The 95% ΔV requirements for the various runs made are summarized in Table 5-VIII.

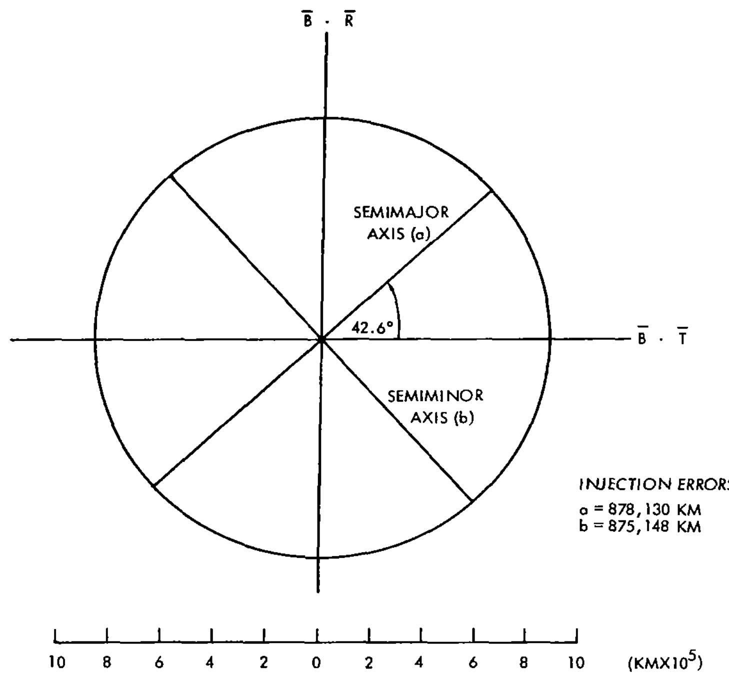

Figure 5-5. Uncorrected Miss Ellipse for Mars Trajectory, Type I

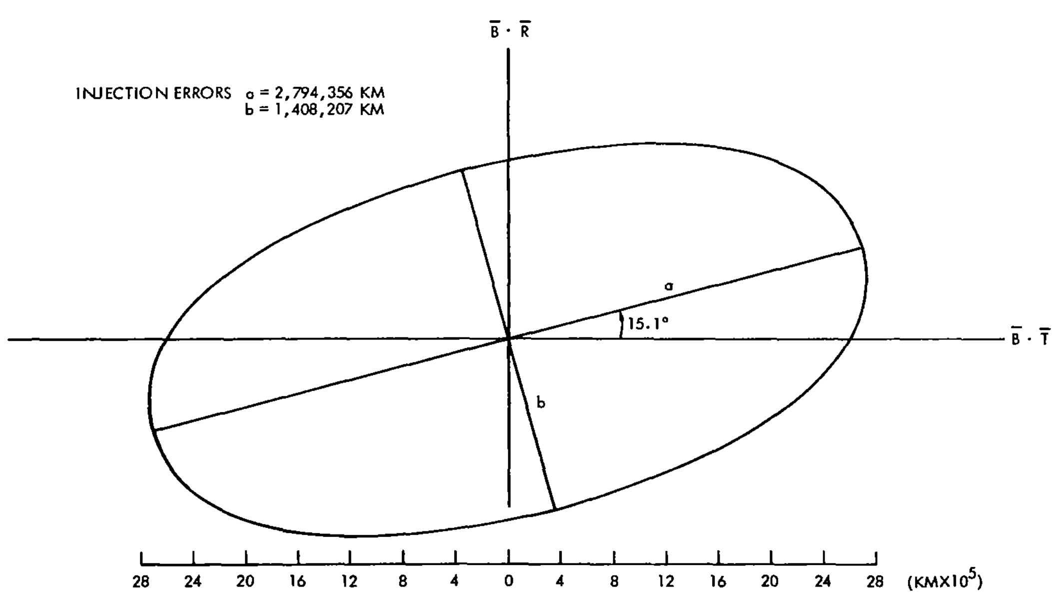

Figure 5-6. Uncorrected Miss Ellipse for Mars Trajectory, Type II

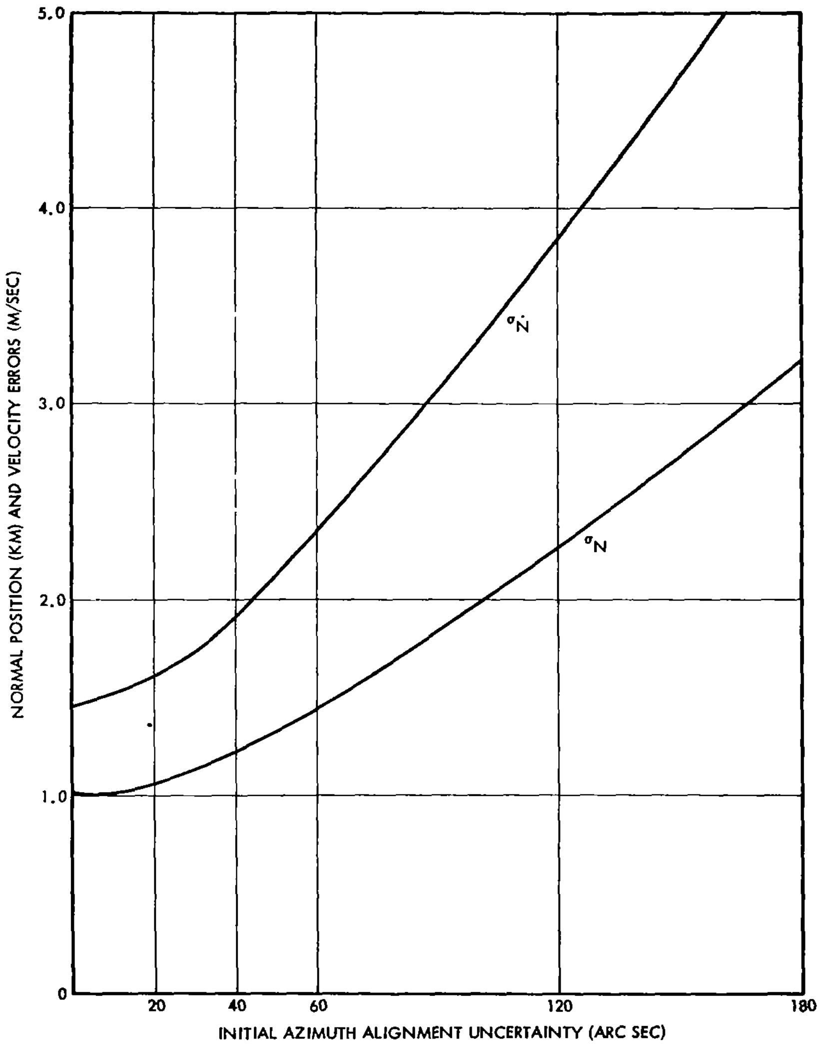

Figure 5-7. Normal Component Sensitivity to Initial Azimuth Uncertainty

TABLE 5-VIII

NINETY-FIVE PERCENT ΔV MIDCOURSE (5 DAYS) REQUIREMENTS FOR 1975 MARS MISSIONS

Trajectory Type	Correction Type	System	95% ΔV (m/sec)
I	M + T*	TG-166	77.8
I	M**	TG-166	70.5
II	M + T	TG-166	77.1
II	M	TG-166	57.6
I	M + T	TG-266	35.4
I	M	TG-266	32.0
II	M + T	TG-266	35.7
II	M	TG-266	26.6

*Miss plus time-of-flight correction.

**Miss correction only.

The Jupiter missions were analyzed for only one value of initial azimuth misalignment, viz, 20 arc sec. For this value, the Saturn IB/Centaur injection errors are those indicated in Table 5-IX for the TG-266 system. The uncorrected miss ellipses at Jupiter due to these errors are illustrated in Figures 5-8 and 5-9. The 95% ΔV required for midcourse correction is summarized in Table 5-X.

TABLE 5-IX

SATURN IB/CENTAUR JUPITER MISSIONS INJECTION ERRORS (RTN COORDINATES)

Position (km)			Velocity (m/sec)		
R	T	N	R	T	N
0.97	1.40	1.89	4.12	2.02	5.46

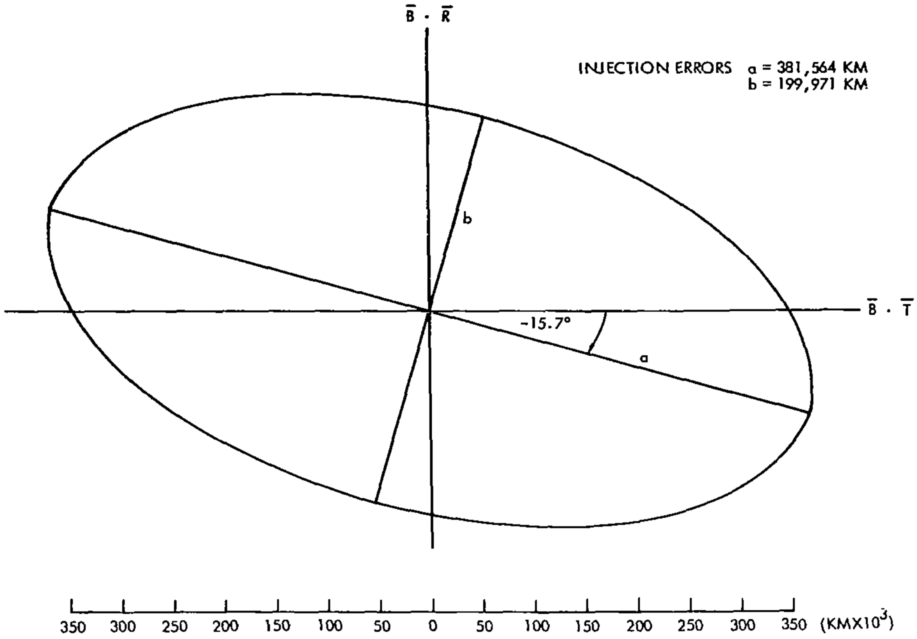

Figure 5-8. Uncorrected Miss Ellipse for Jupiter Swingby/Solar Probe Trajectory

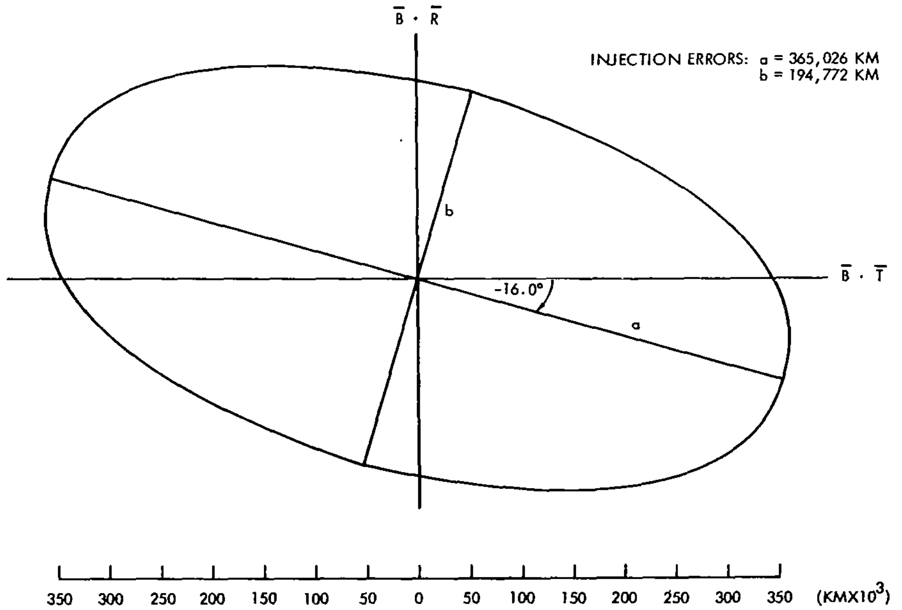

Figure 5-9. Uncorrected Miss Ellipse for Jupiter Swingby/Out of Ecliptic Trajectory

TABLE 5-X

NINETY-FIVE PERCENT ΔV MIDCOURSE (5 DAYS) REQUIREMENTS FOR THE TWO 1971 JUPITER MISSIONS

Mission	Correction Type	95% ΔV (m/sec)
Solar probe	M + T*	10.5
	M**	9.1
Cross ecliptic probe	M + T	10.5
	M	9.1

*Miss plus time-of-flight correction.
**Miss correction only.

5.6 PERFORMANCE ANALYSES FOR THE MIDCOURSE PHASE

Midcourse trajectory corrections are required, in general, to meet the terminal accuracy requirements of lunar and interplanetary missions because for many missions the injection errors, propagated to the target planet or to the moon, exceed the desired errors at encounter. See Figures 5-5, 5-6, 5-8, and 5-9. The injection errors depend somewhat on the launch vehicle characteristics, but primarily on the accuracy of the booster guidance system. The state-of-the-art of boost phase guidance is quite advanced; however, even for the best available guidance systems, the errors at injection considerably exceed those desired for most targeted interplanetary or lunar mission.

The capabilities of ground-based radio tracking and orbit determination techniques (see subsec. 2.2) have advanced to the point where midcourse trajectory corrections can be made with sufficient accuracy to meet the mission terminal objectives with a reasonably small expenditure of spacecraft propellants.

The midcourse correction problem is briefly discussed in this section. A fully attitude-stabilized spacecraft with suitable propulsion for making the necessary maneuvers is assumed.

The guidance concept is similar to that employed in Ranger, Mariner, Surveyor, Lunar Orbiter missions, and other missions:

- The DSIF (S-Band) tracking systems and ground computational facilities are assumed for orbit determination from injection through encounter with the target planet (see subsec. 2.2).
- Based on this determination of the spacecraft position and velocity, corrective maneuvers are computed and transmitted to the spacecraft on-board guidance equipment for execution.

The midcourse maneuver is defined by the impulsive velocity correction, ΔV, necessary to correct the target errors and (optionally) the time of flight.

There are many tradeoffs associated with:

- Single versus multiple midcourse maneuvers and the points at which the corrections are applied
- Allowable spacecraft ΔV capability (this ultimately becomes a tradeoff with payload weight)
- Ranges of possible injection guidance errors (these depend on the booster guidance system and on the launch through injection trajectory)
- Tracking system accuracies attainable (these are a function of the trajectory geometry, tracking radar capabilities and utilization, and ground data reduction capabilities)
- Midcourse maneuver execution errors (these depend on the sophistication of the on-board optical/inertial system)

Analysis of these tradeoffs is beyond the scope of this study.

5.6.1 Midcourse Guidance Techniques

Midcourse guidance is performed by pointing the spacecraft thrust in a direction so that a single velocity increment removes the target errors. This technique, called "arbitrary pointing," was used with Ranger, Mariner, and Surveyor, and allows a single correction to remove all target errors or to remove two components of miss at the target (critical plane correction) and ignore time-of-flight errors.

Target errors are conveniently specified in terms of the components of the impact parameter vector $\overline{B}$ in the R-T plane and the time of flight t_f (see Figure 5-10).

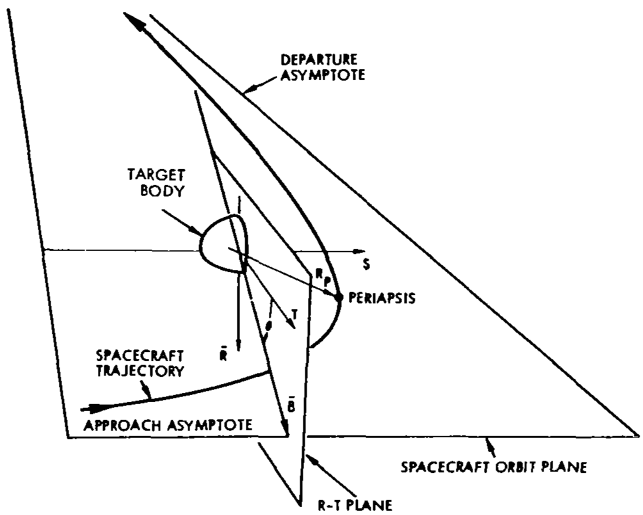

Figure 5-10. Encounter Geometry

For a given interplanetary trajectory, the impact parameter vector $\overline{B}$ specifies in which direction from the planet and what distance the approach asymptote lies. $\overline{B}$ is commonly expressed in components $\overline{B} \cdot \overline{R}$ and $\overline{B} \cdot \overline{T}$, where $\overline{R}$, $\overline{S}$, $\overline{T}$ are a right-hand set of mutually orthogonal unit vectors aligned as follows: $\overline{S}$ is parallel to the planet centered approach asymptote, $\overline{T}$ is parallel to the plane of the ecliptic and positive eastward, and $\overline{R}$ completes the set and has a positive southerly component. The magnitude of $\overline{B}$ determines the distance of closest approach to the planet and the angle

$$\theta = \tan^{-1} \frac{\overline{B} \cdot \overline{R}}{\overline{B} \cdot \overline{T}}$$

specifies the orientation of the planet-centered orbit plane as a rotation about the $\bar{S}$ axis. These definitions are illustrated in Figure 5-10.

5.6.2 Post-Midcourse Trajectory Accuracy Analysis

Estimates for the uncertainty of control of the interplanetary trajectory subsequent to the midcourse correction maneuver are presented in the following paragraphs. The contributions to this uncertainty are the error in execution of the midcourse trajectory correction, the uncertainty in tracking the spacecraft from injection to midcourse correction, ephemeris and astronomical unit errors, and certain identifiable but unpredictable trajectory perturbations acting after the midcourse correction. The midcourse guidance technique described in subsec. 8.2, Ref. 5-1 is assumed for this analysis. It consists of a single midcourse correction about 10 days after launch, with the thrust vector directed essentially parallel to the critical plane to reduce $\bar{B} \cdot \bar{T}$ and $\bar{B} \cdot \bar{R}$ errors.

The root-mean-square and percentage contributions to the target coordinates $\bar{B} \cdot \bar{T}$ and $\bar{B} \cdot \bar{R}$ are listed in Table 5-XI.* The percentage contribution of the total deviation in $\bar{B} \cdot \bar{T}$ and $\bar{B} \cdot \bar{R}$ are computed by assuming that the mean square error contributions are additive.

The midcourse execution errors are calculated for a Mariner-type midcourse guidance system (Configuration Ia described in par. 2.4.1.3, Ref. 5-1) and represent the largest error contribution, as might be expected.

More accurate control of the trajectory, if required, could be obtained by improving the precision of the midcourse maneuver either by using a full strapdown guidance system or by increasing the number of maneuvers. Of the remaining errors, the greatest is the pre-midcourse tracking uncertainty** which causes the estimated position of the spacecraft to be in error. This error is based on present state-of-the-art

*The results in this table were obtained from Ref. 5-5.

**Par. 8.3.1 of Ref. 5-1 describes the results of an analysis of pre-midcourse tracking performed to calculate the state vector uncertainties due to radar tracking and the associated dispersion ellipse at Jupiter. The reader is referred to this paragraph for the detailed results.

TABLE 5-XI

POST-MIDCOURSE TRAJECTORY ERRORS (JUPITER MISSION WITH MARINER TYPE GUIDANCE AND CONTROL) (from Ref. 5-5)

Error Source	RMS $\bar{B} \cdot \bar{T}$ Error (km)	RMS $\bar{B} \cdot \bar{R}$ Error (km)	Percent of Total $\bar{B} \cdot \bar{T}$ Variance	Percent of Total $\bar{B} \cdot \bar{R}$ Variance
Injection errors	951,000	388,000	†	†
Midcourse execution errors††	8,850	10,600	93.0	99.4
Pre-midcourse tracking errors	2,050	625	5.0	0.3
Nongravitation perturbations (unpredictable portions)	1,067	217	1.4	-
Ephemeris errors	500	500	0.3	0.2
Astronomical unit conversion factor uncertainty	303	303	0.2	0.1
Total rss	9,150	10,650	100.0	100.0

Total 99 percent miss ellipse:

Semimajor axis = 26,300 km

Semiminor axis = 17,400 km

†Does not directly affect post-midcourse target errors.

††Arbitrary pointing critical plane correction at 10 days past injection.

tracking accuracies attainable by the DSIF (see subsec. 2.2). Presumably, by 1972 greater accuracy can be attained. Likewise, ephemeris errors and uncertainty in the astronomical unit are based on present state-of-the-art and by 1972 will be appreciably reduced.

5.6.3 Midcourse Execution Errors

Orientation and execution errors introduced by the midcourse correction subsystem have been evaluated for a Mariner-type strapdown guidance system and the TG-166 strapdown inertial guidance system. The results appear in Table 5-XII. It is evident that at least an order-of-magnitude improvement is available by using the more sophisticated strapdown inertial system. Optical sensor accuracies are comparable in the two systems (3-arc min inertial attitude accuracy in each axis is assumed for the latter system).

TABLE 5-XII

COMPARISON OF MIDCOURSE EXECUTION ERRORS FOR TWO TYPES OF INERTIAL GUIDANCE SUBSYSTEM MECHANIZATIONS

	Mariner-Type Simplified Strapdown Guidance System	TG-166 Full Strapdown Guidance System*
Proportional velocity error	0.75% (1σ)	0.043% (1σ)
Pointing error	0.67° (1σ) (11.6×10^{-3} rad)	0.06° (1σ) (10^{-3} rad)
Resolution error	0.0188 m/sec	(Negligible)
ΔV error in performing a maximum 100 m/sec maneuver	0.75 m/sec (1σ) (parallel component) 1.2 m/sec (1σ) (lateral component)	0.04 m/sec (1σ) 0.1 m/sec (1σ)

*See subsec. 7.1 for error model.

The errors presented in Table 5-XII for the two types of optical/ inertial systems may be applied directly to the analysis of the midcourse correction requirements for other missions and to other maneuvers such as orbit insertion. The resultant mission errors will, of course, be different from those given above for the Jupiter mission.

The TG-166 performance satisfies all of the midcourse correction and orbit insertion ΔV requirements summarized in par. 3.3.2.3 (Table 3-III). The TG-266 system, which has better accelerometer performance, also satisfies these requirements. The actual miss ellipses due to the midcourse correction errors are shown in Figures 5-11 through 5-14 for the Mars and Jupiter missions studied in this report.

5.7 NAVIGATION PERFORMANCE ANALYSES FOR PLANET APPROACH PHASES

The radio/optical/inertial tracking and navigation error analyses were conducted using the SVEAD computer program. The results of the study, presented in sec. 9, Ref. 5-1,* are summarized here. Briefly, this error analysis was concerned with the comparative performance of DSIF tracking (earth-based doppler) and onboard optical navigation. Optical instruments considered were: star Canopus sensor, planet (Mars) sensor, and Sun sensor. The planet sensor is used in conjunction with the other sensors to make measurements of the cone and clock angles (defined below) and to make an angular subtense (range measurement) of Mars. Major error sources considered were: slowly drifting biases in the optical equipment, uncertainty in the diameter of Mars, Mars ephemeris errors, doppler-bias error (slowly drifting), and uncertainty in the dynamic model of the solar system (i.e., errors in solar radiation forces on the spacecraft, gravitational constants, planet oblateness, etc.).

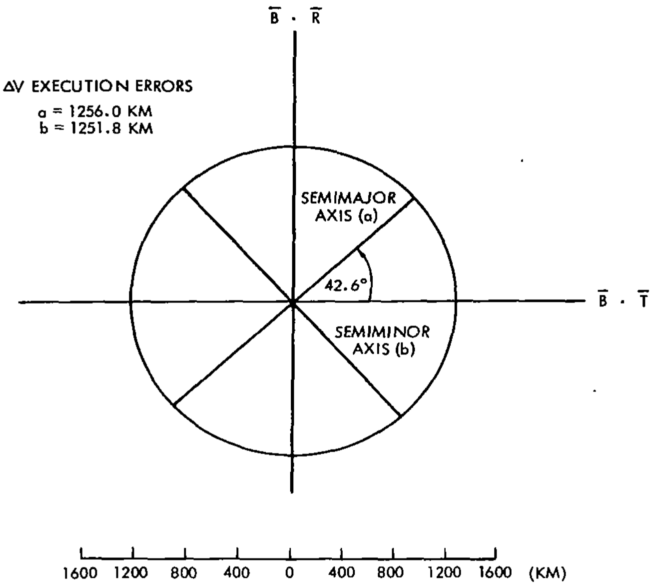

Figure 5-11. Miss Ellipse After First Midcourse Correction for Mars Trajectory, Type I

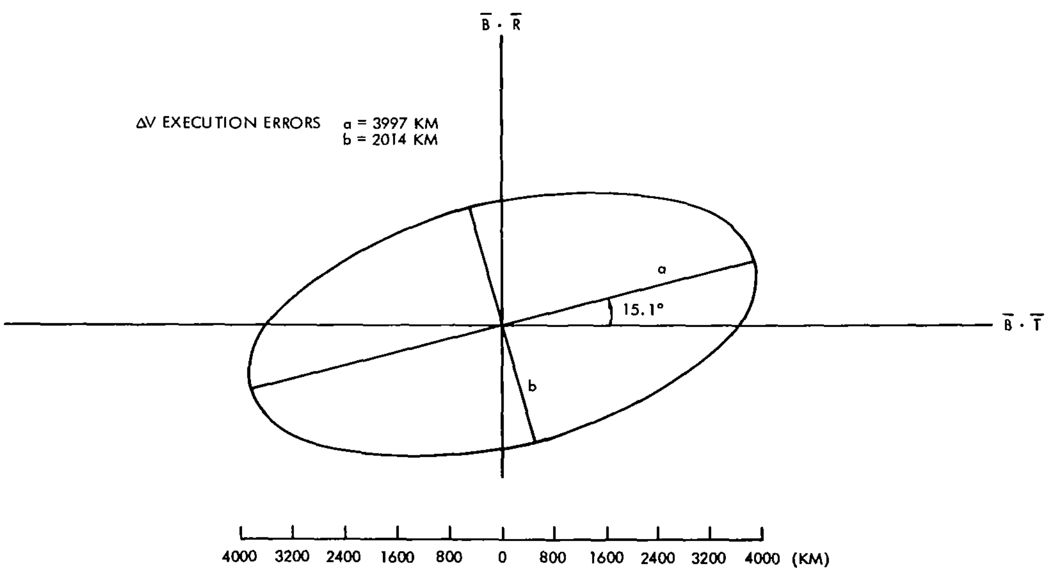

Figure 5-12. Miss Ellipse After First Midcourse Correction for Mars Trajectory, Type II

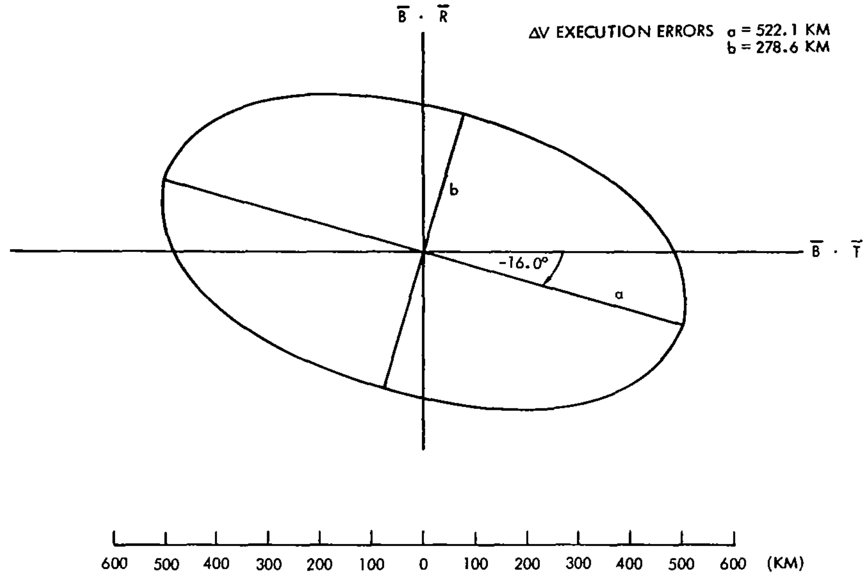

Figure 5-13. Miss Ellipse After First Midcourse Correction for Jupiter Swingby/Out of Ecliptic Trajectory

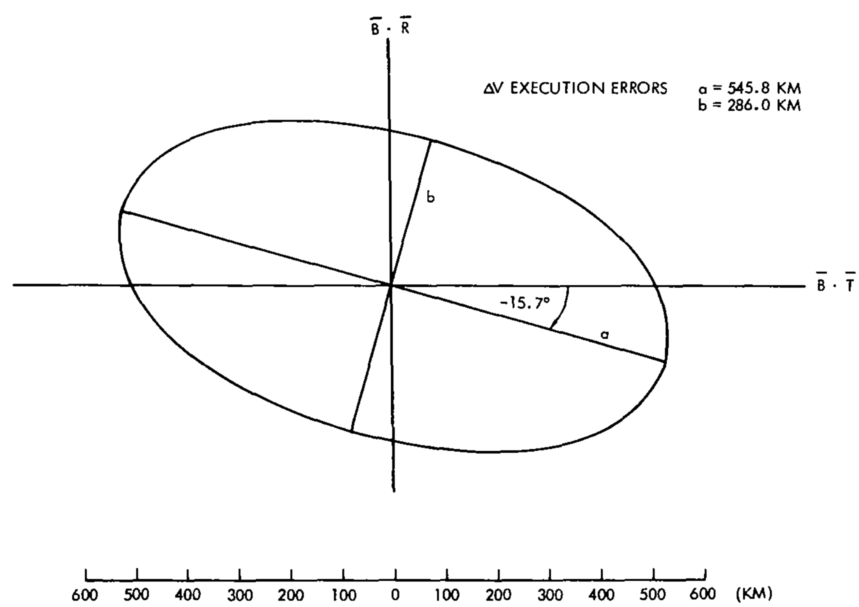

Figure 5-14. Miss Ellipse After First Midcourse Correction for Jupiter Swingby/Solar Probe Trajectory

The principal purpose of the optical measurements is to locate the position of the planet (Mars) relative to the spacecraft. The lines of sight to two known stars may be used to provide a known coordinate system in which Mars may be located. For this study, one star was taken to be Canopus, and the other was taken to be the Sun. Mars is then located by a cone angle ψ and a clock angle θ, as shown in Figure 5-15. The angle ϕ, shown in Figure 5-15, is the Sun-Canopus angle. The subtense angle α, not shown, is an angular diameter measurement which can be used to determine the distance to Mars. Useful optical measurements, for the trajectory considered in this study, could be made over the period from 350 hr to 0.5 day prior to Mars encounter (Mars perifocus).

MEASUREMENT GEOMETRY

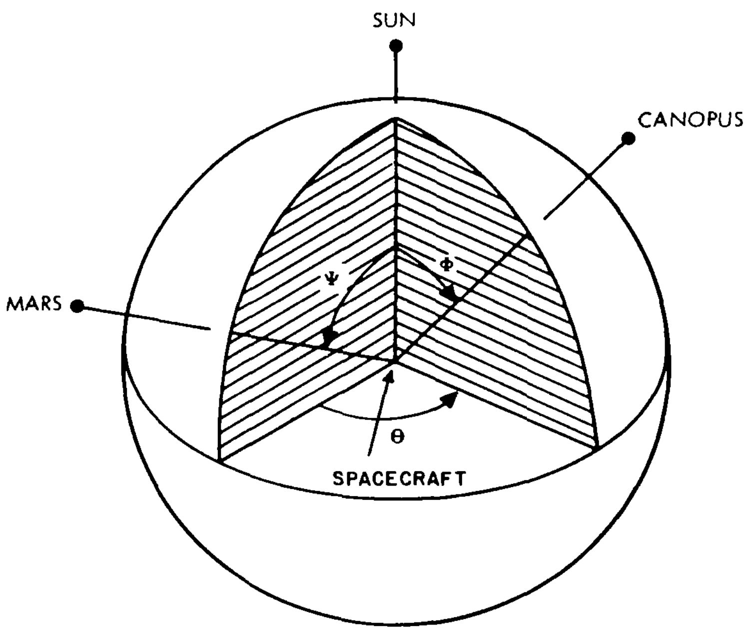

Figure 5-15. Optical Angle Measurements

All "biases" in the error models were assumed to be slowly drifting random variables, exponentially correlated in time. Thus each bias error has a standard deviation and a time constant associated with it; the larger the time constant, the more nearly constant is the bias.

Tables 5-XIII and 5-XIV give the error models used.

Three different error models for the electro-optical sensors were used to investigate the possible improvements in orbit determination accuracies possible by using the Mars approach sensor. Table 5-XIV is the final error model adopted as a result of this study. The other optical error models used are given in subsec. 4.6, vol. II.

Detailed study results of Mars approach orbit determination accuracy and the impact of this accuracy on areocentric orbit insertion and on fuel requirements are presented in subsec. 4.6, vol. II. Based on these results, the following conclusions can be made.

- Stadimetric ranging (comparing Case 4 against Case 5) does not improve overall navigation accuracy.
- The degree to which the addition of optical tracking improves approach orbit determination accuracy over that attainable with doppler only tracking is marginal for in-plane parameters.
- Optical tracking does improve the accuracy to which out-of-plane parameters can be determined (0.2° to 0.5° as against 2° to 5°).
- The deboost velocity requirements to achieve the desired areocentric orbit under ideal conditions (perfect approach orbit determination and perfect execution) is 2.45 km/sec.
- Approach orbit determination errors would lead to improper application of the deboost velocity increment and thus non-nominal areocentric orbits. To make subsequent orbit corrections because of this, the additional velocity penalty would be:

  Case 1—Doppler only 90 m/sec

  Case 4—Doppler plus optical
  Model C, 60 m/sec
- On the basis of difference in velocity requirements, the use of a planetary approach sensor is difficult to justify.

TABLE 5-XIII

RADIO/OPTICAL/INERTIAL ERROR MODEL MARS MISSION

Error	$(\text{Variance})^{1/2}$	Time Constant
Random acceleration acting on spacecraft* (models uncertainty in the dynamic model of the solar system, i.e., errors in solar pressure forces, gravitational constants, etc.)	0.531×10^{-8} m/sec^2 (0.174×10^{-7} ft/sec^2) (causes a 200-km position error in 176 days)	1 week
Tracking system errors		
• Range rate bias	10^{-2} m/sec (0.0328 ft/sec)	1/3 day
• Uncorrelated noise on doppler rate	0.732×10^{-2} m/sec (0.024 ft/sec) (equivalent to 0.12 ft/sec per 1-sec sample, 25 meas. averaged)	
Vehicle errors at injection (3 hr)		
• Position	2 km (6560 ft)	
• Velocity	2 m/sec (6.56 ft/sec)	
Mars ephemeris error (relative to Earth)		
• Position	220 km (7.22×10^{-5} ft)	
• Velocity	0.05 m/sec (0.164 ft/sec)	
Radius of Mars	20 km (6.56×10^{4} ft)	1 day
Uncertainty in gravitational constant of Mars	8.59556 km^3/sec^2	
Uncertainty in second zonal harmonic of Mars	0.48×10^{-3}	

*Equivalent error averaged over 25 measurements.

TABLE 5-XIV

OPTICAL ERROR MODEL C

Error	$(\text{Variance})^{1/2}$		Time Constant
Sun sensor bias	0.407×10^{-3} rad	(1.4 arc min)	1/2 week
Sun sensor uncorrelated noise	0.349×10^{-4} rad* $(0.1746 \times 10^{-3}$ rad)**	(0.12 arc min)	
Mars sensor bias	0.153×10^{-4} rad	33 arc sec	1/2 week
Canopus sensor bias	0.727×10^{-4} rad	15 arc sec	1/2 week
Mars sensor uncorrelated noise	0.349×10^{-4} rad* $(0.1746 \times 10^{-3}$ rad)**		
Canopus sensor uncorrelated noise	0.1746×10^{-4} rad* $(0.873 \times 10^{-4}$ rad)**		
Mars subtense measurement			
Lower limit on $(\text{variance})^{1/2}$ of bias	0.485×10^{-4} rad	(0.17 arc min)	1/2 week
Error proportional to subtense angle	0%		
Uncorrelated noise	0.1745×10^{-4} rad* $(0.873 \times 10^{-4}$ rad)**	0.06 arc min (0.3 arc min)**	

*Equivalent error of 25 measurements averaged. This value was used in the error analysis.

**Single measurement error.

5.8 MARS ORBIT DETERMINATION FROM DSIF TRACKING DATA

The accuracy of orbit determination while the spacecraft is in an areocentric orbit was obtained using the SVEAD computer program (see Ref. 5-1, app. D) for the nominal 1,100 x 10,000-km orbit obtained from the Type I heliocentric transfer orbit. The orbital characteristics are shown in Table 5-XV.

TABLE 5-XV

ORBITAL PARAMETERS FOR MARS ORBIT

Semimajor axis (a)	8960 km
Eccentricity (e)	0.496
Inclination (i)	36.6°
Longitude of ascending node (Ω)	143.1°
Argument of perigee (ω)	-12.3°
Period (T)	7.15 hr

For the 1,100 x 10,000-km orbit obtained, the spacecraft goes behind Mars 9 min after periapsis and is visible again 31 min later.

The initial state vector errors used in the analysis were those obtained at the end of the approach orbit determination phase (doppler tracking only). All other error models were the same as those used in the approach orbit determination phase (Refer to Table 5-XII.

The resulting behavior of the uncertainties in the spacecraft position and velocity in RTN coordinates (see par. 1.4.5) are illustrated in Figures 5-16 and 5-17 for slightly more than one complete orbit. The corresponding orbital elements are illustrated in vol. II Figures 4-33 and 4-34.

The results shown indicate approximately an order of magnitude reduction in the initial errors over a period of one orbit. These results are valid only if no significant local gravity anomalies or other unknown disturbing accelerations are present. The only method of validating this assumption is by analysis of actual tracking data obtained for a spacecraft in Mars orbit.

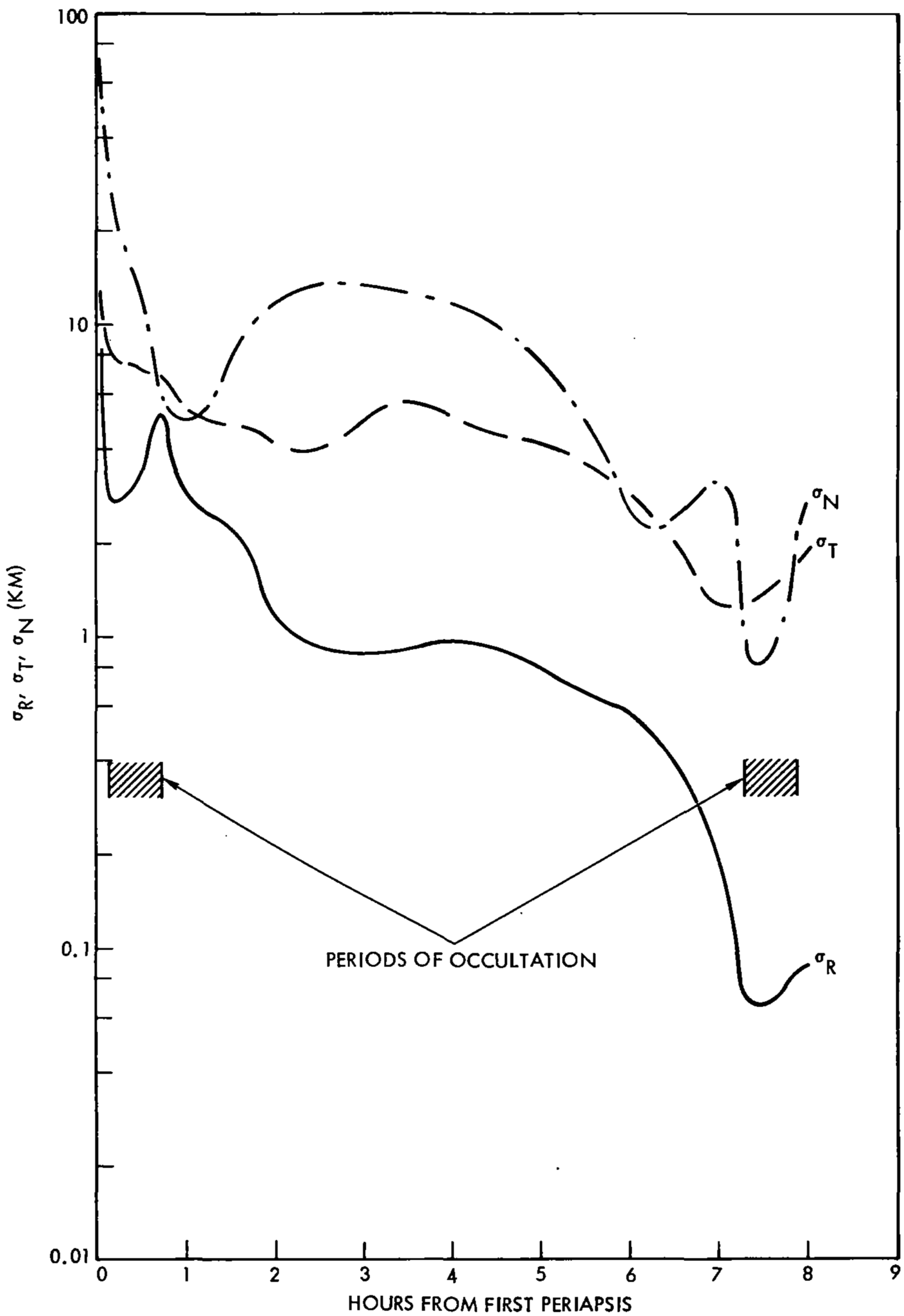

Figure 5-16. Position Uncertainties Versus Time for Spacecraft in Mars Orbit (DSIF Tracking)

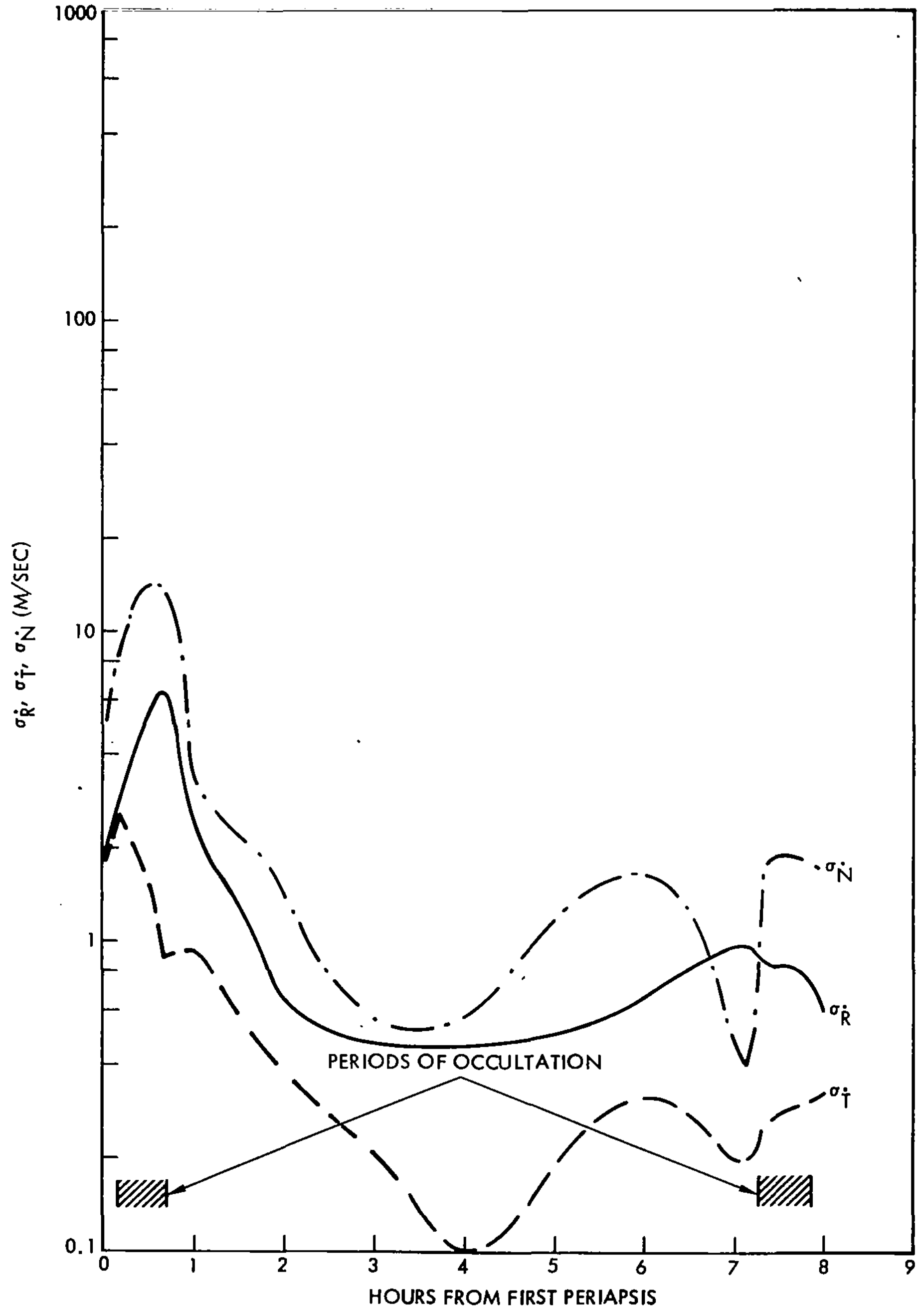

Figure 5-17. Velocity Uncertainties Versus Time for Spacecraft in Mars Orbit (DSIF Tracking)

REFERENCES

5-1 "Radio/Optical/Strapdown Inertial Guidance Study for Advanced Kick-Stage Applications, Volume II; Detailed Study Results," TRW Systems Final Report No. 07398-6017-R000, 30 June 1967.

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5-4 D. Strawther, "Centaur Guidance System Accuracy Report, AC-10 Accuracy Analysis," General Dynamics Report No. GD-C-BTD 64-013-11, 1 April 1966.

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6. SUMMARY OF PRELIMINARY MODULAR DESIGN

6.1 SUBSYSTEM INTERFACES

Preliminary modular designs of radio/optical/inertial guidance, navigation, and control system packages have been configured to meet the functional and performance requirements established for each of the five missions. The basic conceptual design configuration developed for the specified missions, trajectories, and boost vehicles were summarized in sec. 4 of this report. The system interconnections and interfaces of the preliminary modular designs for each of the missions are shown in Figures 6-1 and 6-2. The TG-166 or TG-266 strapdown inertial sensor assembly and guidance computer are central to each of the configurations/missions and provide launch and boost-guidance capability. Computer input and output functions and design of this unit are described in sec. 7, vol II. Each configuration requires a controls electronics interface unit to provide an appropriate interface between the core ROI and the various boost-stage control systems. Boost-vehicle rate gyros are utilized where needed. Preliminary modular design considerations pertinent to the electro-optical sensors, controls subsystem, the onboard computational elements, and the inertial reference unit are discussed in secs. 5, 6, 7 and 8, respectively, of vol. II.

It should be emphasized that in a broad preliminary design study such as this, specific detailed considerations of the total thermal design problem are neither warranted nor defensible. This is particularly true for the lunar and interplanetary missions and is in great part due to lack of knowledge of 1) the types and quantity of other equipment to be installed in the spacecraft and 2) the specific spacecraft thermal design. As a result, much of the thermal design discussions presented in sec. 9, vol. II are general in nature, or as in the case of the IRU, devoted largely to thermal control during the prelaunch and boost periods. The general discussions of the spacecraft/system thermal interfaces are based primarily on past design studies.

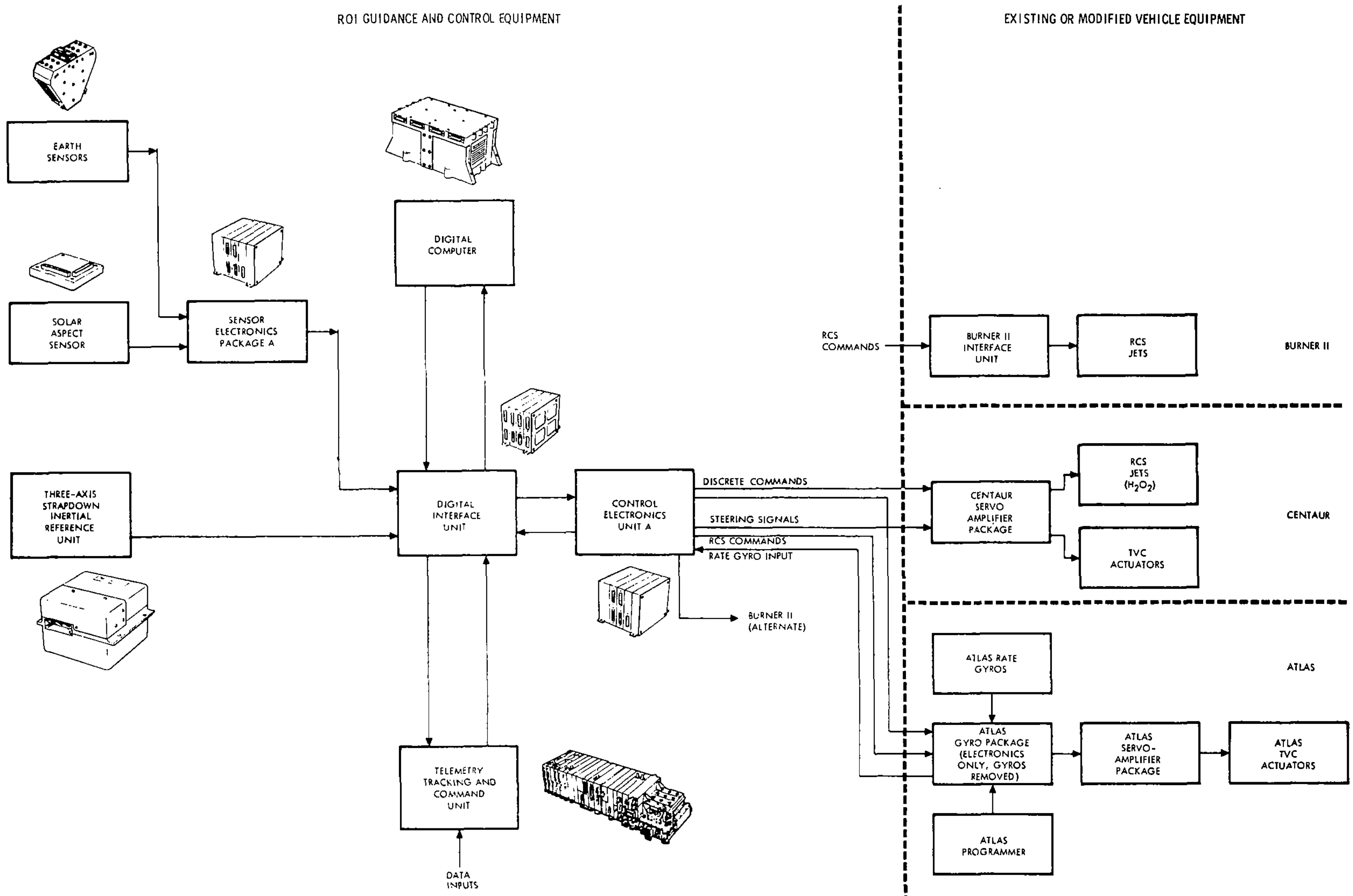

Figure 6-1(a). ROI Equipment Configuration and Interfaces for Earth Orbiting Missions (Atlas/Burner II or Atlas/Centaur)

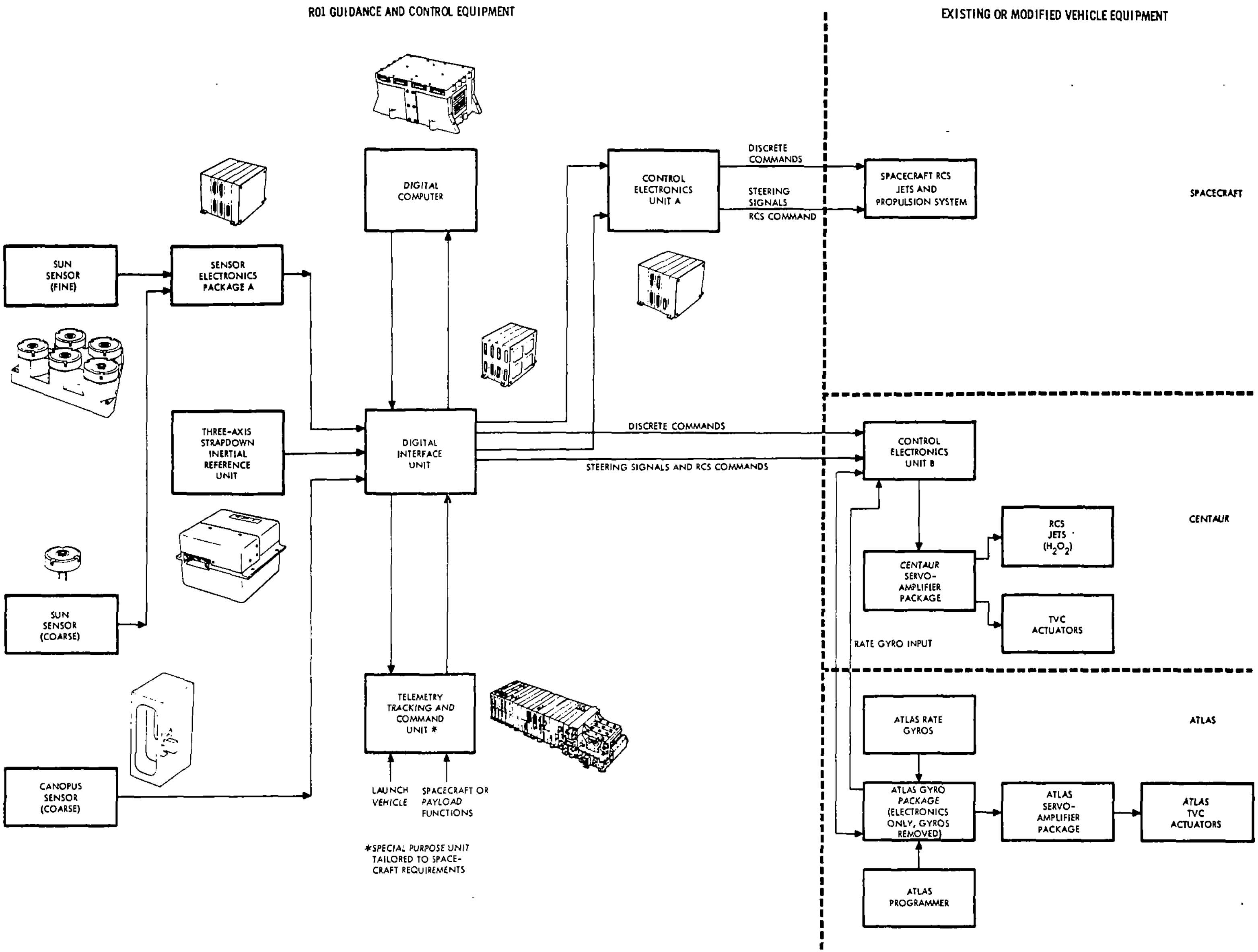

Figure 6-1(b). ROI Equipment Configuration and Interfaces for Lunar Orbit Mission (Atlas/Centaur)

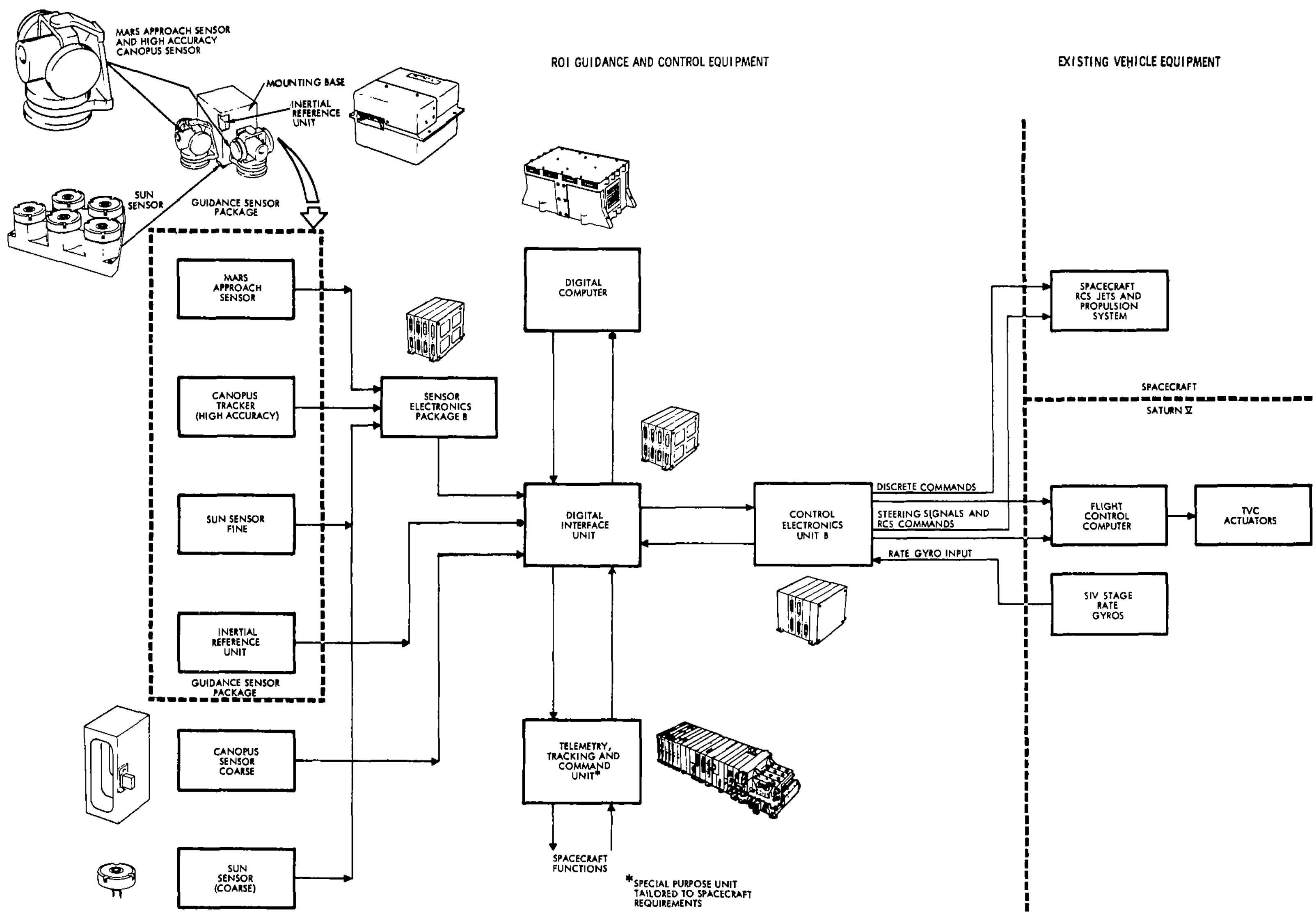

Figure 6-2(a). ROI Equipment Configuration and Interfaces for Mars Orbiter Mission (Saturn V/ Voyager)

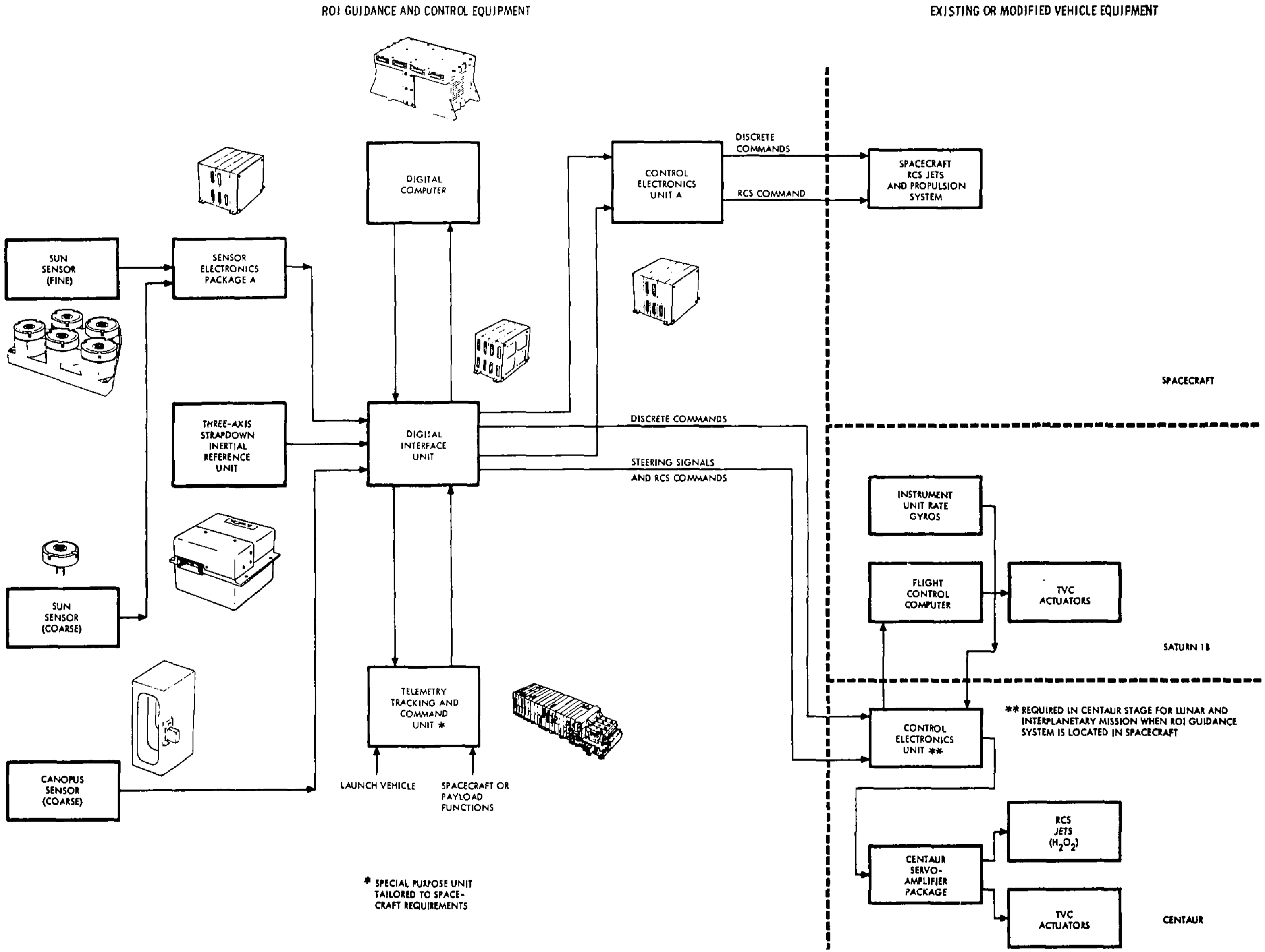

Figure 6-2(b). ROI Equipment Configuration and Interfaces for Solar Probe (Jupiter Swingby) Mission (Saturn 1B/Centaur)

6.2 VEHICLE INTERFACES AND MECHANICAL MOUNTING CONSIDERATIONS

Physical locations of the ROI guidance system components for the five launch vehicle/mission combinations considered in this study are shown in Table 6-I.

TABLE 6-I

ROI EQUIPMENT LOCATION BY MISSION

Mission	Launch Vehicle	Equipment Location
Earth-polar orbit	Atlas/Burner II	Burner II stage
Earth-synchronous orbit	Atlas/Centaur	Centaur stage
Lunar orbiter	Atlas/Centaur	Centaur stage for Surveyor type payload (alternate location in spacecraft)
Solar probe (Jupiter swingby)	Saturn 1B/ Centaur	Interplanetary spacecraft
Mars orbiter	Saturn V	Voyager spacecraft

As was indicated in sec. 2, use of rate gyros located remotely within the Atlas Stage (Atlas/Burner II and Atlas/Centaur) or within the Saturn IV (Saturn 1B/Centaur) or the S-IVB Stage (Saturn V-Voyager S/C) is required. In addition, for those missions utilizing the Centaur stage and where the ROI system is located within the spacecraft but is providing down-stage guidance functions, a controls electronics package is required in the Centaur stage in addition to the controls electronics package in the spacecraft.

Equipment locations in the Burner II locations are illustrated in Figure 6-3. Equipment is mounted on the spacecraft structure in the locations shown (typical). There are no critical mounting requirements.

For the Centaur stage (earth-synchronous orbit mission) the ROI guidance equipment is mounted on the forward end of the Centaur stage on a mounting shelf provided for that purpose. Figure 6-4 shows a typical mounting arrangement. Optical sensors and the IRU must be located in

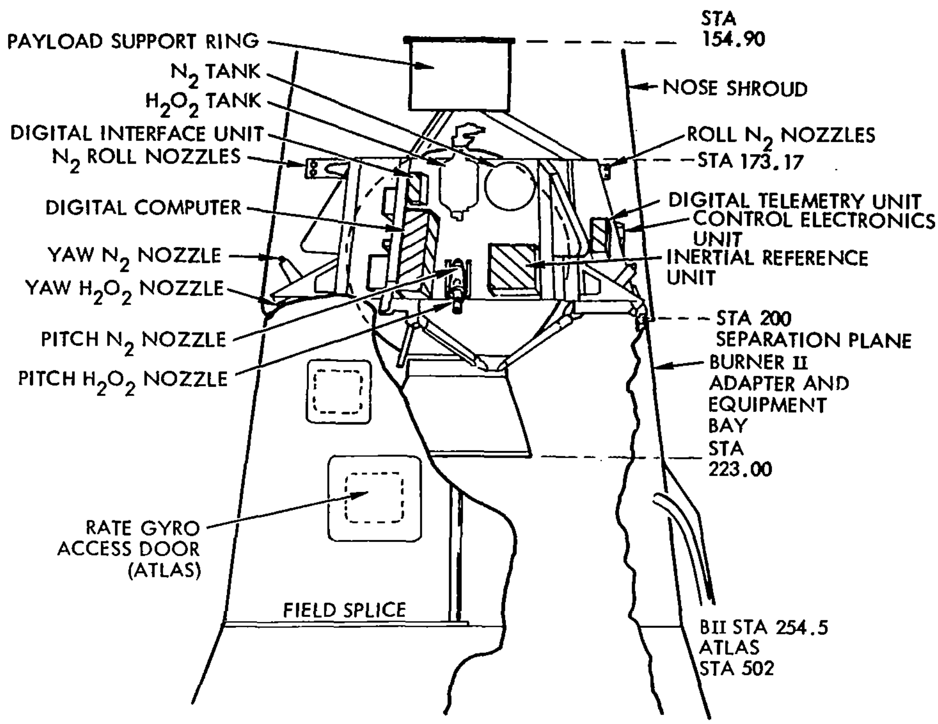

Figure 6-3. Equipment Location and Mounting Within the Burner II Stage

close proximity on a base that provides a moderate degree of alignment stability between the instruments throughout the mission. In addition, an unobstructed line-of-sight to the earth and sun must be provided at those mission times (prior to perigee and apogee burns) when attitude or time updates are required. In general, this will require attitude maneuvers of the Centaur stage during the coast period prior to the apogee burn.

Spacecraft location of the ROI guidance equipment is required for the Mars orbiter mission and for other missions where the ROI system provides the spacecraft cruise attitude control and guidance functions for powered maneuvers. Equipment location within the Voyager spacecraft is illustrated in Figure 6-5. For this mission, sensor alignment tolerances are critical during the Mars approach phase. Use of a precision navigation base is recommended which mounts the optical instruments used for navigation measurements during the approach phase (see subsec. 6.3).

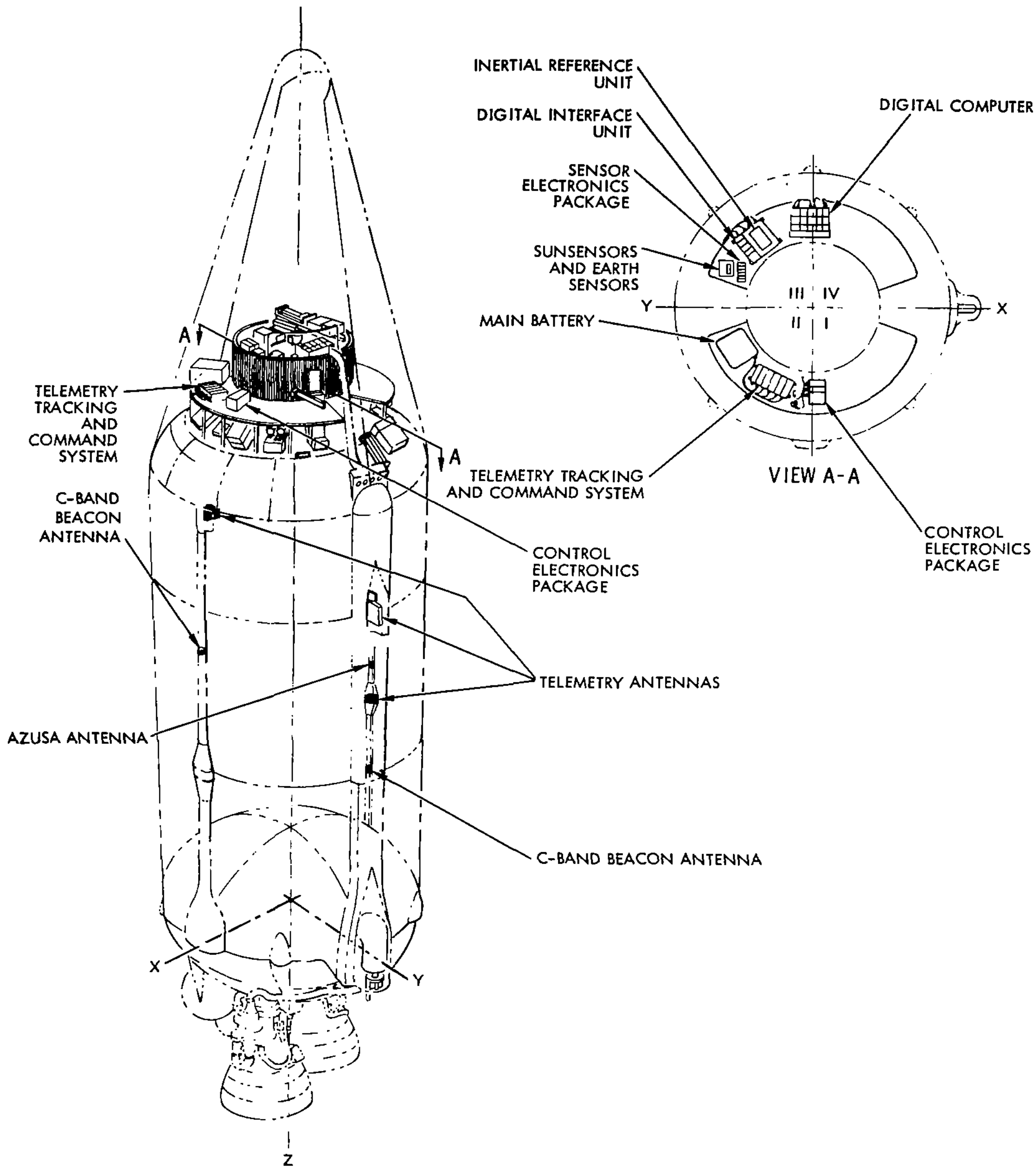

Figure 6-4. Equipment Location and Mounting Within the Centaur Stage

Unobstructed line-of-sight to the Sun, Canopus, and Mars must be provided during the last 10 days prior to encounter. The guidance sensor package, consisting of the navigation base assembly, the Mars approach

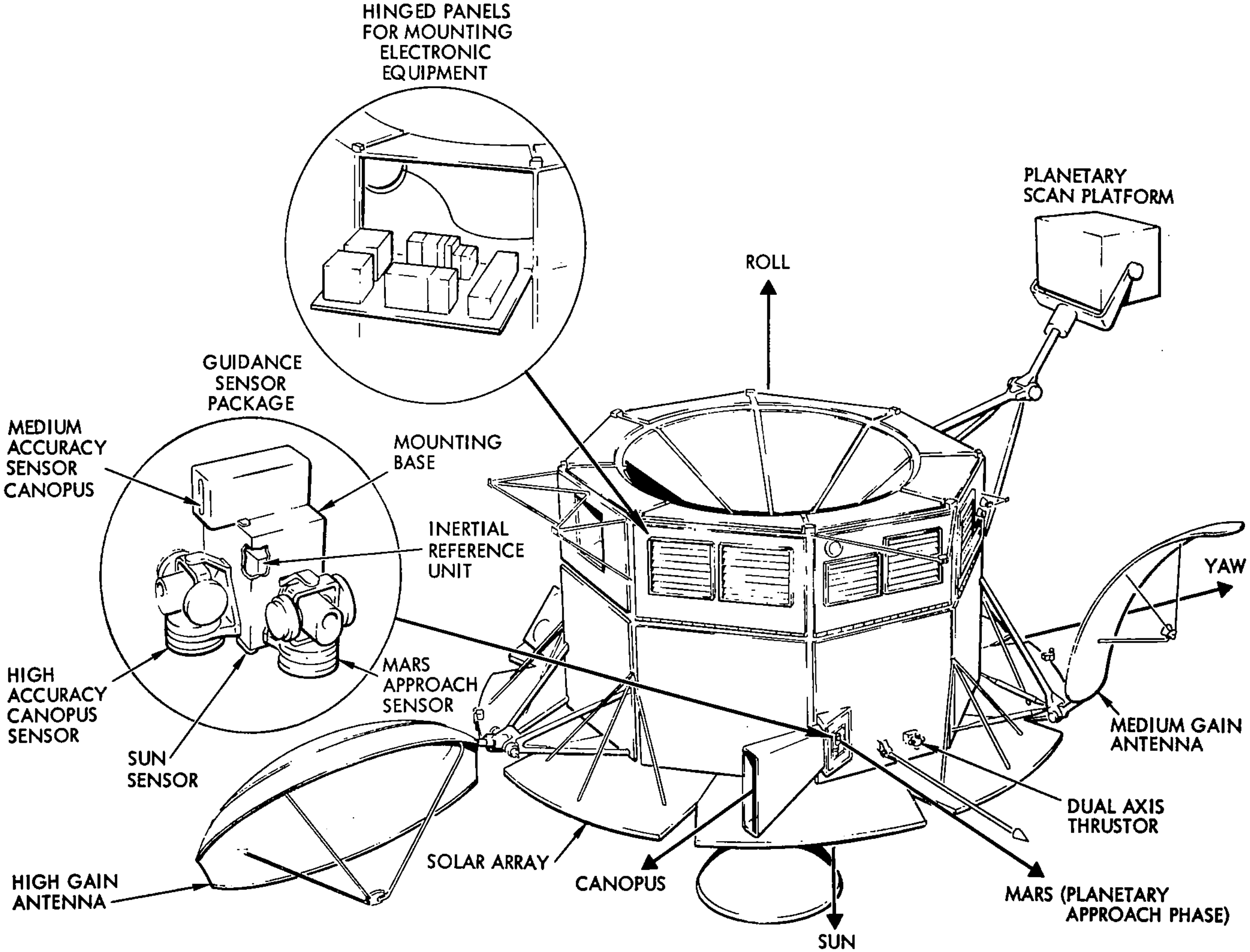

Figure 6-5. Equipment Location and Mounting Within the Voyager Spacecraft

sensor, the precision Canopus tracker, and the IRU, is mounted as a unit to the spacecraft structure as illustrated in Figure 6-5. Electronic packages are mounted to the hinged panels as shown.

6.3 GUIDANCE EQUIPMENT MECHANICAL INTERFACE AND PACKAGING CONSIDERATIONS

6.3.1 Navigation Platform Subassembly

The electro-optical sensors which require critical alignment and the IRU are mounted on a common base or navigation platform to minimize the effects of vehicle flexure on relative alignment accuracy. Each electro-optical sensor is a separate module, mechanically and functionally independent. Various modules are mounted on the navigation platform, or elsewhere on the vehicle structure, to make up a guidance system configured to the requirements of a specific mission. The mounting base is designed both to accommodate the sensors required and to be structurally compatible with the spacecraft or launch vehicle selected for the mission. The platform is positioned on the vehicle so that the fields-of-view of the optical sensor are not obstructed. Thermal control shielding must also allow an unobstructed view for the sensors as well as provide the required temperature control.

A navigation platform subassembly, as described above, is required for mounting and aligning the approach guidance sensors and the IRU for Mars orbiter mission and for accurate referencing of the sun sensor and earth sensor to the IRU for the earth-synchronous orbit mission. The sensors required for cruise attitude control in the Mars and lunar orbiter and solar probe (Jupiter swingby) missions do not require such precise referencing and may be mounted directly to the spacecraft structure in accordance with standard spacecraft design practices.

6.3.2 Sensor Alignment

The sensors must be accurately aligned to the vehicle axes to provide valid data for guidance purposes, considering the limits of their pointing capabilities. The coarse sun sensors do not require accurate alignment in relation to the spacecraft, but they do require an unobstructed view. Therefore, regardless of the orientation of the spacecraft, a signal can be provided to orient the vehicle to bring the sun into the field-of-view of the fine sun sensor. Four coarse sun sensor eyes (Ball Bros. CE-5) are mounted around the periphery of the spacecraft where they have an unobstructed field-of-view, pointed so there will be no dead zone. They will be paired in relation to the vehicle axes with one pair controlling the pitch axis, the other pair the yaw axis. Except during earth or planetary eclipse, the sun will always be in the field-of-view of at least one of the sun sensors.

Alignment requirements for the other optical sensors are as follows:

Kollsman KS-203 Canopus tracker	±5 sec
NASA/ERC planetary approach tracker	±5 sec
Adcole 1402 digital aspect sun sensor	±15 sec
BBRC Fe-5A fine sun sensor	±30 sec
TRW 246164 earth sensor	±3 min
Honeywell Mars '69 Canopus tracker	±3 min

Optical sensors from this group that are required for a specific mission will be mounted on the navigation platform.

The most critical alignment requirements are for the KS-203 star tracker and the planetary approach tracker. The star tracker must be provided with precision-machined mounting surfaces which interface with its mounting on the navigation platform. The mounting surfaces on the tracker housing are the references for all alignment processes and accuracy tests of the tracker and can be aligned during its manufacture. A similar mounting and alignment arrangement can be used for the planetary approach tracker.

The use of a boresight alignment fixture is recommended for alignment of the navigation sensor assembly to the spacecraft or launch vehicle. The mounting surfaces of this fixture are identical to those on the tracker and can be attached to the star tracker mounting surfaces on the navigation platform. The boresight fixture has two auto-collimating surfaces (mirrors) from which errors can be measured between the x, y, and z axes of the vehicle and the corresponding axes of the tracker mount. With the boresight fixture in place, the platform is aligned to the spacecraft. After alignment, the boresight fixture is removed and the KS-203 star tracker installed.

During manufacture of the navigation platform, the star tracker mount would be machined first. It will then be used as a reference for machining all the other sensor mounts. Manufacturing tolerances must be well within the alignment requirements for the other sensors. For the Mars orbiter mission where both the KS-203 star tracker and the planetary approach tracker are used, the mounting interface for the planetary approach tracker would be hand-lapped to the required accuracy in relation to the star tracker mount. The boresight fixture and optical tooling equipment can be used to assure that the required accuracy is achieved. Correct alignment of the KS-203 star tracker mount will thus assure that all the sensors on the platform are properly aligned. On missions which do not require the KS-203 star tracker, the mount for the most accurate sensor in the group would be used as a reference for machining the other sensor mounts and for referencing to the IRU reference porro prism.

This mounting technique permits not only the selection and mounting of just the sensor modules required for a specific mission, but also provides for removal and replacement of a faulty sensor in the package without requiring realignment of the unit. A possible sensor package configuration for the Mars mission is illustrated in Figure 6-6.

6.4 ONBOARD COMPUTATIONAL ELEMENTS

As part of the definition of the preliminary modular design of the radio/optical/strapdown inertial guidance and control system, studies were conducted to obtain the functional sizing of the computer and to establish both the computer interface and the preliminary requirements for modular design. The results of these studies are summarized below.

The general characteristics of the computer were assumed to be those defined in the preliminary specification of the NASA-ERC/UAC Advanced Kickstage Guidance Computer (Ref. 6-1). This computer is composed of two essentially independent computers configured from two memory units, two arithmetic units, and two control units. The computer is described in sec. 7, vol. II.

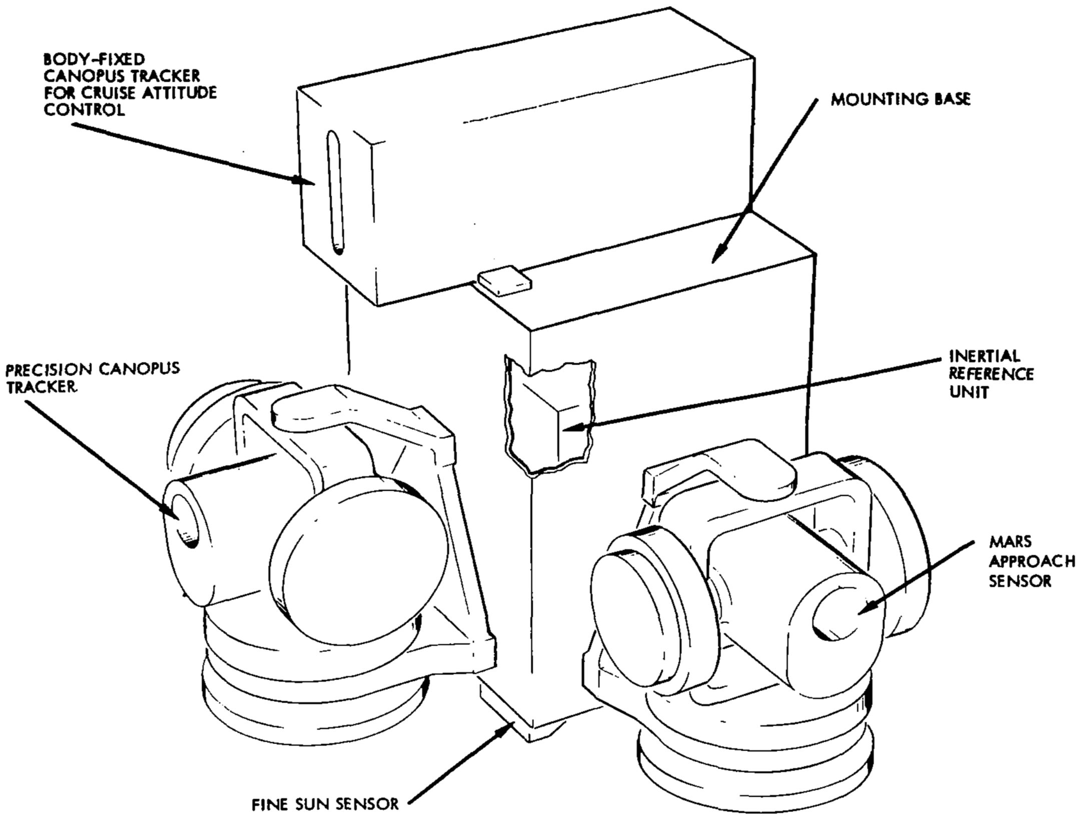

Figure 6-6. Guidance Sensor Configuration for the Mars Mission

Estimates of computer memory size and computational speed were based, substantially, on past computer experience gained from such TRW projects as the computer design for the LM Abort Guidance System and Centaur advanced guidance studies (Refs. 6-2, 6-3, 6-4, and 6-5). Although the memory size and speed requirements are expected to be reasonably accurate, they have not been completely verified by trial programming and simulation.

The second part of this study subtask, the refinement of the computer interface and the preliminary modular design, emphasized detailed definition of the logical and functional I/O interfaces among the computer and the guidance, control, command, and tracking subsystems. Additionally, the estimated reliability of the TRW-designed computer interface unit was based on a nonredundant design and the use of off-the-shelf integrated circuits. A reliability estimate also was made for the ERC/UAC Advanced Kickstage Guidance Computer, based on parts failure rate data supplied by ERC.

6.4.1 Computational Requirements

The major guidance and control functions performed by the computer are shown diagramatically in Figures 4-1 through 4-5 of this volume and are listed as follows:

1) Prelaunch computations and initialization
2) Direction cosine computation algorithm
3) Coordinate transformation and navigation computation
4) Euler angle and angular rate computation
5) Ground tracking network input
6) Output telemetry routine
7) Guidance and steering laws
8) Thrust vector and reaction jet control laws
9) Navigation time update (long parking coast earth-synchronous mission only).

Estimates were made for the computational timing and data and program storage requirements of the major functions outlined in the previous paragraph. The method of making these estimates is discussed in par. 7.2.3 in vol. II. The directional cosine computation, coordinate transformation and velocity transformation, Euler angle and angular rate computations, and guidance and steering laws are based on the LM Abort Guidance System equations as programmed on the NASA-ERC/UAC Advanced Kickstage Computer. The timing and storage requirements for the thrust vector control and reaction jet control laws are based on the equations shown in Figures 7-3 through 7-7, vol. II. The estimates of timing and storage requirements for the guidance and steering equations are based on the explicit guidance equations contained in Ref. 6-5.

An additional 25% of the timing and storage is added to estimates to account for miscellaneous factors, e.g., scaling, intermediate steps, deviations of equations, and uncertainties in computer characteristics.

The estimated storage and time requirements are summarized in Table 6-II. The most stringent, single-precision word length requirement is 29 bits plus sign for the strapdown attitude algorithm (see par. 7.2.2, vol. II).

6.4.2 Computer Interface Unit

The conceptual design of the computer I/O unit and its interface with the various ROI components are summarized in this subsection.

The configuration of the equipment must meet the composite requirements for all missions so that specific components can be interchangeably combined into effective operational systems. A key to making this concept practical is the achievement of simplified component interfaces to avoid unnecessary excess capability and the resulting penalty in weight and power consumption. In addition, having component interfaces compatible with a GP computer I/O unit facilitates accommodation of any combination of these components. In this manner, mission-dependent changes may be accomplished by suitably modifying the stored computer program.

In the recommended preliminary modular design all I/O operations of the computer occur via a computer interface unit (CIU) that contains provisions for communication with any combination of auxiliary sensors.

The CIU contains the hardware elements that link the guidance computer to the other ROI components and auxiliary sensors. It performs pulse accumulation, format conversion, control decoding, buffer storage, and generation and conditioning of command and control signals. The elements are organized by functions for parallel access to a computer input or output channel. Figure 6-7 shows a block diagram of the CIU which indicates the functional relationship of the internal elements and ROI system.

CIU interfaces with the inertial reference unit, the various electro-optical sensors, and other elements of the modular guidance and control systems that are defined and described in detail in vol. II, sec 7.

TABLE 6-II

ESTIMATES OF TIMING AND STORAGE REQUIREMENTS

Functions	Storage Requirements	Timing Requirement (msec)
Prelaunch computations and initialization	194	--
Direction cosine computation algorithm	203	6.2
Coordinate transformation and navigation computation	250	1.3
Euler angle and angular rate computations	68	0.9
Ground tracking network input	70	0.7
Output telemetry	68	0.7
Guidance and steering laws	1226	
• Coordinate transformations and sensed acceleration computations		
• Engine discretes		
• Fixed conic and velocity to be gained computations		
• Atmospheric steering		
Thrust vector and reaction jet control laws	810	9.3
Navigation time update (earth-synchronous satellite mission)		
Sun-local vertical angle	151	5.0
Sun direction	77	
TOTAL	3097	24.1

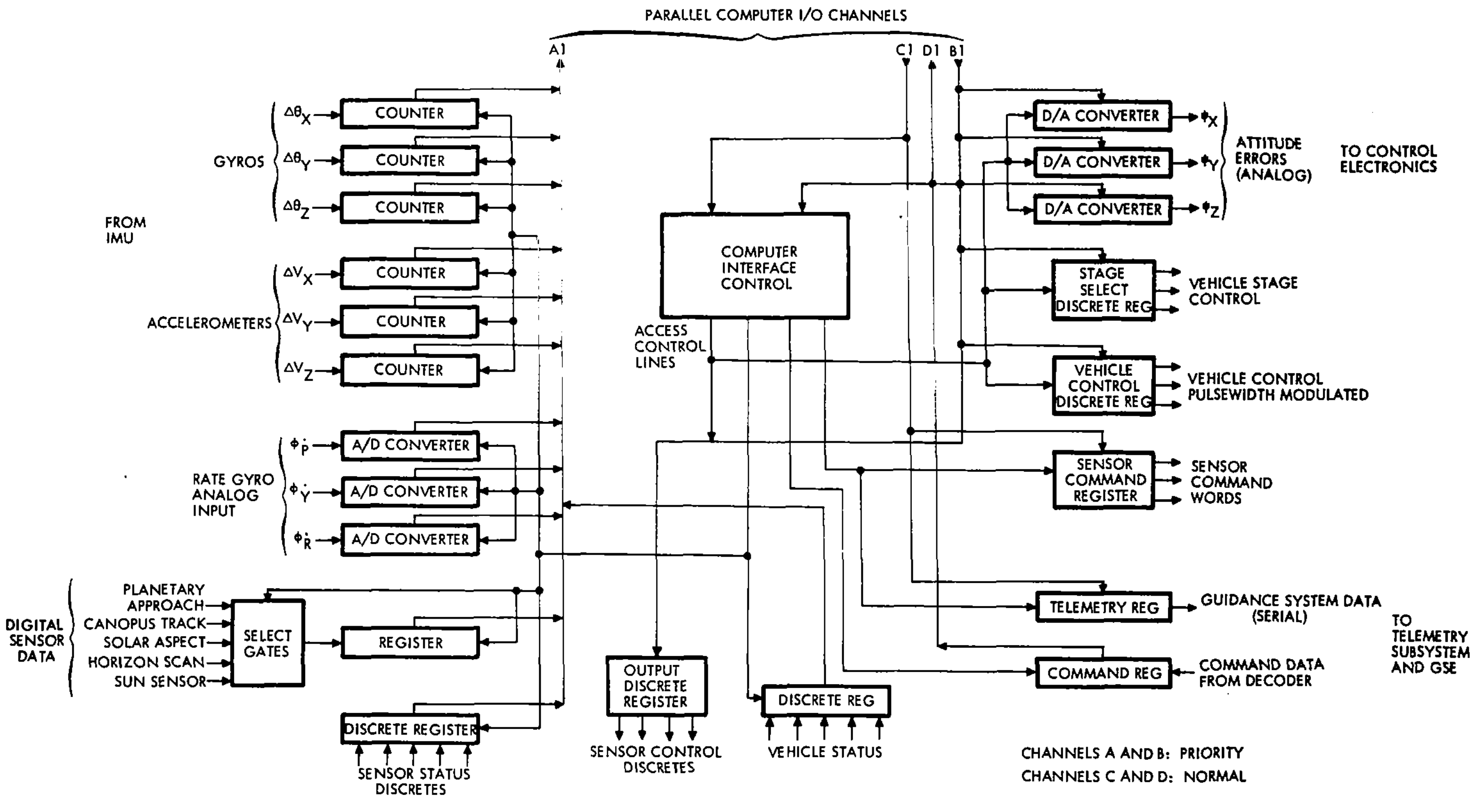

Figure 6-7. Functional Block Diagram of the Computer Interface Unit

6.5 CONTROL SUBSYSTEM MODULAR DESIGN

6.5.1 Summary of Design Concepts

The control subsystem conceptual design summarized in sec. 4 formulates a control system sufficiently general to encompass a variety of selected missions, including the use of a variety of boost vehicles and spacecraft. The generality in design is desired to preclude hardware changes to the Radio/Optical/Strapdown Inertial Guidance System (ROI) and, in particular, to the digital computer. To achieve this, the control system computer requirements must be broad enough to deal with the most complex mission. In addition, the interfaces with the remaining control system equipment must be defined and satisfied without requiring major modifications to this equipment.

A digital control system was selected for the modular design since it provides a flexible means of implementing the control functions on a per flight basis without requiring hardware modifications. In this study, full advantage was taken of the digital computer existing within the ROI guidance system to accomplish the stabilization and control of the boost vehicle as well as the spacecraft. With the use of the computer, a single autopilot can be employed to control all the booster and spacecraft stages thereby eliminating the need for the multiple autopilots, which often are used in multistage space boosters. Other benefits of this approach include 1) the elimination and/or simplification of certain items of booster control hardware and 2) the ease of the ROI guidance and control system in adapting to the various booster/spacecraft thrust vector control (TVC) and reaction control systems (RCS) with a minimum of special purpose interface hardware.

It is also feasible to mechanize a single set of control equations within the flight computer to control each of the booster and spacecraft powered flight phases, including midcourse corrections and planetary deboost phases (as well as coast flight phases) through programmed changes in autopilot gains and filter coefficients. The computer equations can be modularly programmed such that only the needed portion of the equation set is used for control during each mission phase. This reduces the computational requirements and is particularly attractive during the coast phases since it enables the computer to give more attention to other aspects of the mission, e.g., experiment control and data handling.

In addition to modularity in the equation software, similar modularity in control system hardware can be achieved with different electronic packages fabricated to interface with the different boost vehicles. The design of the interfacing packages is highly dependent on the degree of modification acceptable for each boost vehicle.

Several control system design configurations were considered for the selected space boosters, varying from minimum to maximum modification of the existing control system electronics. The recommended designs are essentially intermediate modifications in which the existing downstate rate gyros are employed with output signals routed upstage to the computer via an electronics package which provides interface compatibility. Use of downstage rate gyros was found to be necessary for the Atlas/Centaur and Saturn V vehicles and desirable for the Saturn IB/ Centaur vehicle. In past missile and booster designs, location of these gyros on the boost vehicle generally has been necessary, particularly in the more flexible vehicles.

A new control electronics package for the Centaur vehicle is recommended which would replace the present programmer and gyro packages and would interface with the ROI guidance system. The Centaur sequencing functions would require initiation by the computer. The 400-Hz signal modulation and demodulation function would also be performed within this package to satisfy the booster signal interface.

A computer interface with two control electronics packages, one for the spacecraft and one for the upper stage, is considered desirable to produce a versatile design configuration while minimizing the spacecraft electronics package weight. The signal mixing for differential roll TVC would be performed within these packages as well as power amplification of discretes, stage selection, servoamplification, engine signal biasing, and signal conditioning to satisfy telemetry and booster interface requirements.

The minimum modification control system designs generally require proper placement of the rate gyros and frequent design changes of the electronic filters to accommodate changes in payload; hence, they may cost more. On the other hand, filter changes within the ROI guidance system are accomplished simply through program coefficient changes and add little to programming cost.

As a preliminary estimate, the number of filter coefficients, gains, limits, deadzones, and other autopilot parameters required are expected to be less than 364, assuming a three-stage boost vehicle plus a spacecraft on a complex mission such as a Mars orbit mission. This number assumes separate pitch and yaw filters. The control autopilot sampling period of 25 samples/sec is considered satisfactory for the powered flight phases, and the computational delay of 10 msec assumed in the performance analysis was acceptable, although delays less than 5 msec would be desirable. The estimated number of required autopilot parameters included a seventh-order digital filtering capability (three quadratic modules plus an integrator). In the performance analysis, a fifth-order filter or less was needed with proper placement of the rate gyros on the booster. Use of the ROI guidance system as proposed is shown to produce good control system designs from marginal ones because of its multiple-gain and filter-change capabilities.

The detailed control equations are given in Volume II, subsec. 8.3.

6.5.3 Control System Interfaces

The control equation flow and interface with control electronics packages is indicated in Figure 6-8. The computer interface is maintained as simple as possible to preclude modifications due to booster and spacecraft changes. The signal mixing operation for differential roll control is performed within the electronic package as are the stage selection operations. This enables simplification of the thrust vector command interface to three command lines for each of the two electronic packages, or a total requirement of six analog output channels. The

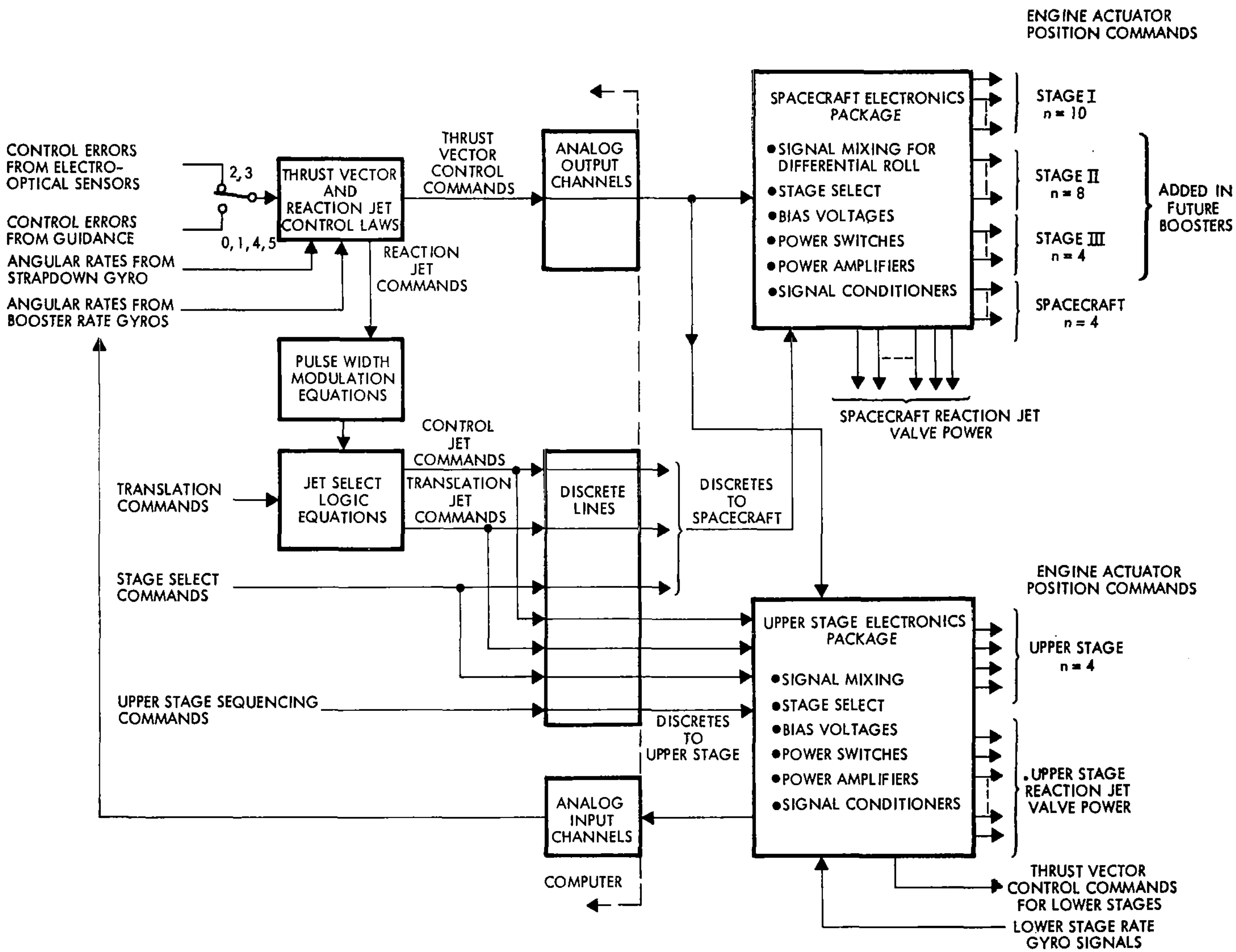

Figure 6-8. Control System Signal and Equation Flow Diagram

routing of the command signals to the appropriate operating stage would be accomplished through discrete commands issued to the electronic logic circuit.

The spacecraft electronics package shown in the figure indicates the possibility of furnishing thrust vector commands to all stages for future booster designs. This will be particularly attractive for solid propellant boosters since minimal stage sequencing is required as opposed to liquid propellant vehicles which require considerable sequencing of the propellant supply system. In this type of unified design, the number of thrust vector command lines may be as few as 6 and as many as 26; hence, the use of electronic signal mixing and stage selection to maintain the computer interface intact would be desirable. Moreover, 26 analog channels would represent a costly requirement on the computer design.

Also in the figure, the three analog input channels for the downstage rate gyro signals are indicated. It is expected that only one set of booster rate gyros will be required; however, if additional gyros on different stages were used, they could also be selected with the stage select discretes and logic.

In addition to the functions mentioned, the electronics packages would provide power switches for execution of the discrete commands, power amplification of the thruster vector command signals, summation of command signals with actuator feedback signals, 400-Hz modulation and demodulation for signal compatibility with downstage electronics, signal conditioning of telemetered signals, electrical bias voltages for required engine canting, and regulated voltages and current for the electronic circuits.

The use of two electronic packages, one for an upper stage and one for the spacecraft as shown in Figure 6-8. presents a highly flexible design configuration. In vehicles where passive or autonomous spacecraft are employed, the ROI system could be mounted on the upper stage and the upper stage electronics package employed. In vehicles where an upper stage is not added and an active spacecraft is employed, the upper stage electronics package and the ROI system could be mounted within the spacecraft, thereby retaining the interface with the booster providing that the package weight is acceptable. Use of both packages would occur in vehicles which contain an added upper stage and an active spacecraft.

It is recommended that the ROI guidance system modularity be maintained through use of different control electronics packages rather than a single one. The use of a single upper stage electronics package to interface with all candidate boost vehicles is not recommended since such a design is expected to be heavier, more costly, and more difficult to design. Moreover, impedance matching electronic circuitry for the numerous output lines needlessly dissipates electrical power.

6.6 SUMMARY OF PERFORMANCE CHARACTERISTICS

The overall performance characteristics of the preliminary modular Radio/Optical/Strapdown Inertial Guidance System design are summarized in the following paragraphs.

6.6.1 Trajectory Accuracy and Fuel Required for Correction of Guidance, Navigation, and Control Errors

Position and velocity errors at orbit injection are summarized in Table 6-III for the five missions considered in this study. The table also gives the 95% ΔV required to correct the injection errors by orbit trim or midcourse maneuvers and the target miss errors due to the midcourse execution errors, assuming a single midcourse correction is made at injection plus 5 days.

For the lunar and interplanetary missions, the tracking accuracy available from DSIF is such that, at midcourse, the initial target miss due to injection errors is effectively corrected. Significant trajectory errors subsequent to midcourse are due to midcourse execution errors and spacecraft accelerations due to unpredictable forces (e.g., uncertainties in solar pressure effects), uncertainties in the knowledge of gravity fields, and target planet ephemeris uncertainties. Further tracking by DSIF

TABLE 6-III

T6-166 AND T6-266 INJECTION AND MIDCOURSE PERFORMANCE SUMMARIES

Mission	Position and Velocity Component Errors at Injection (in RTN Coordinates)				95% ΔV Requirements to Correct for Injection Errors (m/sec)		Target Miss Due to Midcourse Execution Error (One Midcourse at Injection + Five Days)	
	TG-166		TG-266					
	R/T/N (km)	$\dot{R}/\dot{T}/\dot{N}$ (m/sec)	R/T/N (km)	$\dot{R}/\dot{T}/\dot{N}$ (m/sec)	TG-166	TG-266	$\overline{B}\cdot\overline{R}$ (km)	$\overline{B}\cdot\overline{T}$ (km)
Near-earth polar orbit	1.28/1.59/4.01	3.28/1.50/6.05	0.54/0.96/1.32	1.56/0.86/1.79	17	6	NA	NA
Earth-synchronous orbit								
a) Direct ascent	50.0/20.8/19.9	6.2/1.8/1.6	30.3/20.3/14.1	5.2/1.1/1.2	9	8	NA	NA
b) Parking orbit	59.5/148.2/84.3	10.4/2.0/2.1	33.6/142.2/77.6	8.7/1.3/1.8	23	20	NA	NA
Mars injection								
a) Type I	7.1/2.01/3.37	7.96/2.89/4.84	3.37/1.02/1.07	3.52/1.30/1.61	77.8[1] 70.5[2]	35.4[1] 32.0[2]	1250	1250
b) Type II					77.1[1] 57.6[2]	35.7[1] 26.6[2]	3600	2000
Jupiter injection								
a) Solar probe			0.97/1.40/1.89	4.12/2.02/5.46		10.5[1] 9.1[2]	500	290
b) Cross-ecliptic probe						10.5[1] 9.1[2]	520	310

NA = Not applicable
(1) = Fixed time of arrival midcourse correction
(2) = Variable time of arrival midcourse correction

reduces these trajectory uncertainties to tolerable values for all the missions studied with the single exception of the Mars mission. For this mission, initiating the use of the onboard approach guidance sensors 10 days prior to encounter significantly reduces the trajectory uncertainties with respect to the target planet. Plots of the trajectory uncertainties during the approach guidance phase are presented in sec. 5, vol. II.

6.6.2 Summary of System Reliability, Weight, and Power Requirements

Table 6-IV summarizes the weight, power, and total failure rate estimates for each of the elements comprising the modular Radio/Optical/Strapdown Inertial Guidance and Control system. Table 6-V summarizes the overall system reliability, weight, and power for each of the five missions considered in this study.

Some assumptions made in computing the mission reliability are listed below:

1) No redundancy has been assumed except that inherent in existing designs.

2) For the interplanetary missions, it is assumed that the inertial reference unit and digital computer are shut down during the interplanetary cruise phase, but are reactivated when necessary to perform powered maneuvers. Zero failure rate during shutdown has been assumed.

3) For the interplanetary missions, it is assumed that course-attitude control is maintained using the Sun sensor and body-fixed Canopus sensor as attitude references. Simple analog control electronics would be utilized, bypassing the digital computer.

4) For the interplanetary missions, it is assumed that the tracking, telemetry, and command (TT and C) system required for guidance purposes is integrated with the spacecraft TT and C subsystem to provide a single system. The configuration of this equipment will be highly dependent on the mission characteristics and data requirements. The design will, therefore, be unique to each mission. For these reasons, system weight, power, and reliability estimates given in Table 6-V do not include the TT and C contribution.

TABLE 6-IV

SUBSYSTEM WEIGHT, POWER, AND FAILURE RATE SUMMARY

		Weight (kg)		Power (W)		Failure Rate/ 10^6 Hr
Inertial reference unit		8.7		72		250
Digital interface unit		3.6		40		52
Digital computer		13.7(3)		60(3)		100
Tracking, telemetry and command		10.6		47		80
Earth sensors		5.0		(1)		0.2(2)
Solar aspect sensor		0.3		(1)		0.4
Sensor electronics A						
Earth sensor electronics	2.3		10.0		2.6	
Solar aspect sensor electronics	0.7		0.8		9.0	
Coarse sun sensor electronics	0.1		0.3		1.0	
		3.1		11.1		12.6
Sun sensor (coarse and fine)		0.23		----		0.09
Canopus sensor (body-fixed)		3.65		5.5		6.5(4)
Mars approach sensor (gimbaled)		9.6		(1)		2.1
Canopus tracker (gimbaled)		9.6		(1)		2.1
Sensor electronics B						
Canopus tracker electronics	4.1		16		40.6	
Approach sensor electronics	4.1		18		39.0	
Coarse/fine sun sensors	0.2		0.5		2.0	
		8.4		24.5		81.6
Control electronics A and B		9.1(5)		30(5)		100(5)
Precision mounting base		8.0		----		----

Notes:

(1) Power included in electronics.

(2) Equivalent Failure rate for three out of four heads operating.

(3) No packaging design exists for the NASA-ERC/UAC computer. The size and weight were estimated by TRW based on rough comparisons with current computer designs.

(4) Reliability data not available from manufacturer. Estimate based on TRW design of a similar instrument proposed for Voyager.

(5) Estimates made by TRW based on similar equipment.

TABLE 6-V

SYSTEM RELIABILITY, WEIGHT, AND POWER SUMMARY

Mission	Mission Duration		System Reliability (Nonredundant System) (Partial Shutdown During Oribital Cruise)	Total System Weight (kg)	Power Requirements	
	Prelaunch[1] and Powered Phases (hr)	Cruise Phases (hr)			Maximum Power (W)	Total Energy (kW-hr)
Earth-polar orbit	20	-	0.988	45.7	250	0.1
Earth-synchronous orbit	38	-	0.998	54.1	260	4.7
Lunar orbit	24	300	0.984[2]	49.1[2]	218[2]	10.0[2]
Mars orbiter	240[3] 385[4]	8,500	0.730[2]	74.7[2](5)	252[2]	150[2]
Solar probe (Jupiter swingby)	23	24,000	0.715[2]	39.1[2]	218[2]	390[2]

Notes:

(1) 20 hr prelaunch operation assumed
(2) Not including tracking, telemetry and command system
(3) Assumed operation time for approach sensors
(4) Assumed operating times for IRU, computer
(5) Includes navigation base assembly

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6-4. "Advanced Centaur Computer Requirements," TRW Systems Report No. 4222-6031-RU-000, May 1965.

6-5. "Centaur Explicit Guidance Equations Study, Final Report," TRW Systems Report No. 08768-6002-R000, 17 January 1967.

7. SUMMARY OF SENSOR PERFORMANCE AND DESIGN CHARACTERISTICS

This section presents a summary of the performance and design characteristics of the strapdown inertial reference unit and electro-optical sensors selected for the modular design. Secs. 10 and 11 of vol. II present detailed descriptions, operating characteristics, performance analyses, and performance specifications for each of the electro-optical sensors and for the inertial reference unit. An interface design of the sensing elements with the digital computer is described in sec. 7 of vol. II.

7.1 STRAPDOWN INERTIAL REFERENCE UNIT

7.1.1 Design Characteristics and Instrument Selection

Based upon the inertial equipment survey presented in Ref. 7-1, vol. III, two representative strapdown inertial reference units (IRUs) were configured for purposes of this study. These IRU mechanizations, denoted by TG-166 and TG-266, are based on presently available inertial instruments and represent a range of readily achievable performance capabilities. Characteristics of the selected IRUs are shown in Table 7-1.

TABLE 7-I

INERTIAL INSTRUMENT SELECTION AND PHYSICAL CHARACTERISTICS OF THE TG-166 AND TG-266 INERTIAL REFERENCE UNITS

IRU Model No.	Gyros	Accelerometers	Volume (cm^3)	Weight (kg)	Power (W)
TG-166	Nortronics GIK7	Kearfott Model C 702401-005	8,200	8.7	72
TG-266	Honeywell GG334	See Ref. 7-1 (vol. IV) for selected accelerometer	11,000	13.0	83

*In order to permit an unclassified presentation of performance data in this section, the identification of the TG-266 accelerometer is made in the Classified Annex, vol. IV of Ref. 7-1.

Although a particular choice of instruments was made for purposes of this study, it is not intended that this choice constitutes a recommendation for development of IRUs based on these instruments. The major motivation for choosing these particular instruments was 1) to span the range of currently available performance capabilities and 2) to select instruments on which a reasonable amount of test data was available to TRW for the purpose of constructing error models.

The TG-166 is an IRU with moderate performance (accuracy) and is available at moderate cost. The TG-266 represents a higher performance IRU subsystem and is available at a higher cost.

The strapdown configuration for both candidate IRUs consists of three single-degree-of-freedom gyros and three accelerometers mounted in an orthogonal triad. A functional block diagram of the TG-166 and TG-266 IRU mechanizations is shown in Figure 7-1. Both mechanizations

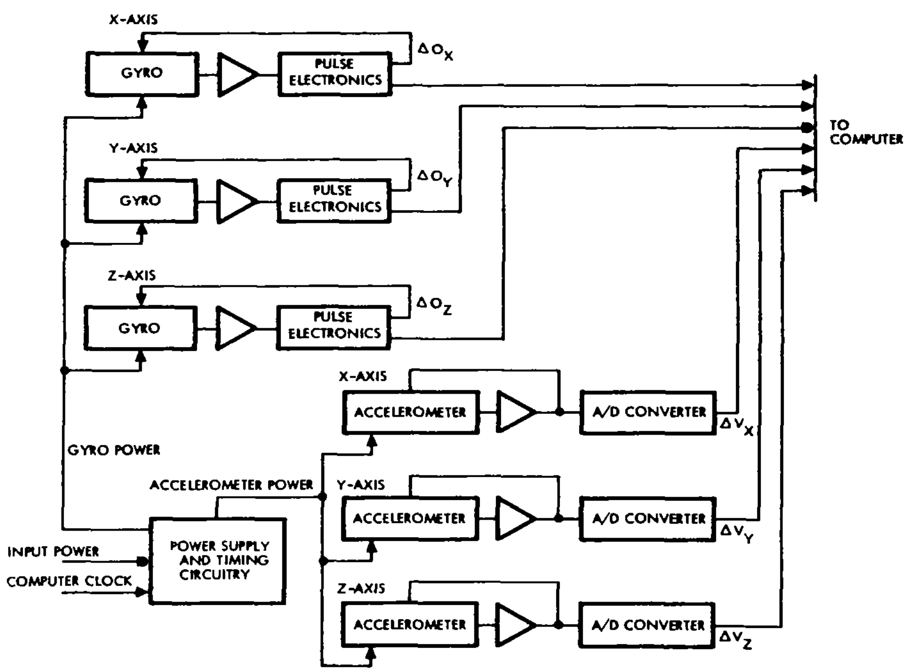

Figure 7-1. TG-166 and TG-266 System Block Diagram

employ pulse torqued* gyros and analog rebalanced accelerometers with analog-to-digital converters providing an interface with the digital computer.

The actual system and loop configurations of the two IRUs are the same except that the TG-266 accelerometer loop utilizes a servo position amplifier instead of a force-to-balance loop.

7.1.2 Performance Characteristics

Error models for the two IRU configurations are summarized in Table 7-II. A detailed discussion and derivation of the error models is given in sec. 4, vol. II, of Ref. 7-1. Figure 7-2 shows the instrument axis orientations assumed.

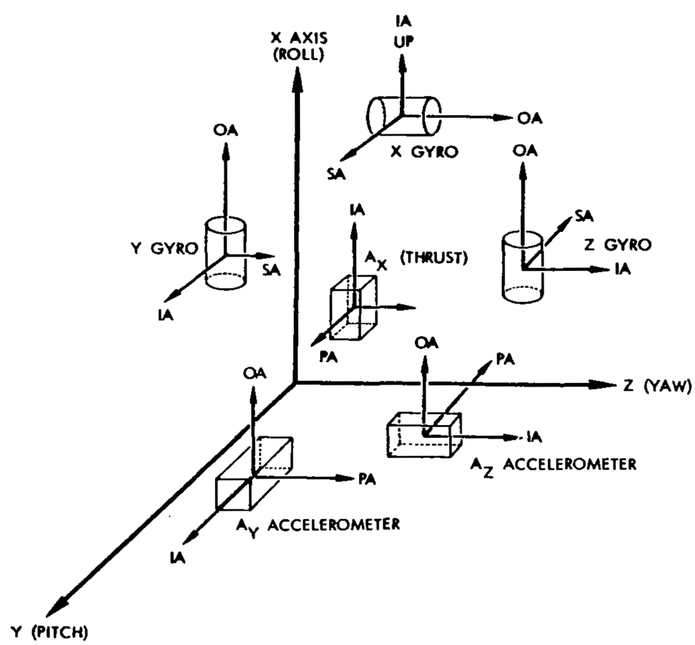

Figure 7-2. Strapdown Coordinate Axes (Prelaunch Orientation)

*Pulse torquing techniques are discussed in detail in app. A to vol. II of Ref. 7-1.

TABLE 7-II

SUMMARY OF PERFORMANCE CHARACTERISTICS FOR STRAPDOWN INERTIAL REFERENCE UNITS TG-166 AND TG-266

Description	TG-166	TG-266	Units
Accelerometer			
Bias	21	14	μg
Scale factor	75	24	$\mu g/g$
x acc. input axis rotation toward y axis	12	10	arc sec
x acc. input axis rotation toward z axis	12	10	arc sec
y acc. input axis rotation toward z axis	12	10	arc sec
Pendulous axis g sensitivity	15	10	$\mu g/g$
Output axis g sensitivity	1	1	$\mu g/g$
Input-pendulous g product sensitivity	50	30	$\mu g/g^2$
Input-output g product sensitivity	0. 5	0. 5	$\mu g/g^2$
Gyro			
Bias drift	0.187	0. 09	deg/hr
Input axis g sensitive drift	0. 627	0. 16	deg/hr/g
Spin axis g sensitive drift	0. 627	0. 16	deg/hr/g
Output axis g sensitive drift	0. 02	0	deg/hr/g
Anisoelastic drift	0. 04	0. 04	$deg/hr/g^2$
Scale factor	57	26	ppm
Gyro input axis rotations toward each of other two axes	10	10	arc sec
Gyro input axis rotations toward each of other two axes	10	10	arc sec

The error model coefficients were derived from hardware sensitivities presented in subsec. 4.3, vol. II, Ref. 7-1. These sensitivities were derived from actual test data, information obtained from the instrument manufacturers, and TRW circuit design studies. In those cases where data were not available, an attempt was made to estimate the error sensitivity terms in a conservative fashion. Although several terms of the error model had to be obtained by this method, the sensitivities which were estimated are generally insignificant in practice.

Two additional error models are presented in Ref. 7-1 for each configuration, one in which a calibration update is performed just prior to launch and one without an update. This correction or updating would be made to the thrust axis accelerometer bias and scale factor and the roll axis gyro fixed drift and mass unbalance along the spin axis within a few hours of launch. The calibration update is derived from a measurement of the output of the thrust accelerometer and roll gyro immediately before or after the system is installed in the launch vehicle and a second measurement just prior to flight. It is shown in Ref. 7-1 that the system statistical figure of merit can thereby be improved.

7.2 ELECTRO-OPTICAL SENSORS

7.2.1 Sensor Selection and Utilization

The method of implementation which has been considered in this study is that of a strapdown inertial guidance system in which the electro-optical sensors are used to update system alignment and bound the errors due to gyro drift. In addition, the electro-optical sensors may be used for regaining control of spacecraft attitude after a complete power shutdown during an interplanetary coast phase or after recovery from a complete power failure.

The candidate electro-optical sensors which have been selected are based upon those defined in a state-of-the-art survey presented in vol. III, Ref. 7-1. Information in this survey was obtained either directly from manufacturers and research laboratories or was extracted from applicable data compiled under the USAF Standardized Space Guidance System Study

(Ref. 7-2). Both the current state-of-the-art and projected advancements were defined in the survey and the following types of optical sensors were included:

- Sun sensors, including both nulling devices and solar aspect sensors
- Earth sensors, including both horizon sensors for use in earth orbit and long-range earth sensors for use in interplanetary flight
- Star trackers, including both gimbaled and strapdown subsystems using both mechanical and electronic scanning, and photoelectric or solid-state optical radiation detectors
- Star field sensors, using photoelectric and solid-state detectors with either mechanical or electronic scanning techniques
- Planet sensors for terminal approach or planetary orbit, employing both mechanical and electronic scanning.

It was determined that the specified missions could be accomplished utilizing various combinations of sun sensors, earth sensors, a Canopus sensor, and a planetary approach sensor. Only in the case of the Mars orbiter mission was it determined that state-of-the-art equipment was not applicable. In this case it was determined that, in order to obtain a higher degree of accuracy,* higher precision would be required for both the Canopus sensor and the planetary approach sensor.

The following paragraphs summarize the operational sequence of utilization of the selected electro-optical sensors for several phases of specified missions. The sensors which have been selected for the various missions are defined in Table 7-III. The performance and design characteristics of the various sensors and a preliminary design concept for

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*This type of mission can be performed with reasonable accuracy without the use of an approach guidance sensor. More specifically, the early Voyager missions can be accomplished using a combination of an onboard optical inertial system (without the approach sensor) with precision earth-based tracking if the projected improvements in the DSIF can be achieved (see subsec. 2.2). Nevertheless, the accuracy improvement due to use of the approach guidance sensor may be useful for advanced orbiter missions.

TABLE 7-III

RECOMMENDED ELECTRO-OPTICAL SENSORS FOR VARIOUS MISSIONS

Applicable Electro-Optical Sensor	Recommended Type	Earth-Synchronous Orbit (Direct Ascent)		Earth-Synchronous Orbit (Parking Orbit Injection)			Earth Polar Orbit		Interplanetary Missions					
										Cruise and Midcourse				
		Launch Guidance	Injection Into Synchronous Orbit	Launch Guidance	Injection Into Transfer Orbit	Injection Into Synchronous Orbit	Launch Guidance	Injection Into Orbit	Injection Into Interplanetary Orbit	Cruise Attitude Control	Midcourse Corrections	Mars Approach	Mars Orbit Injection	Lunar Orbit Injection
<u>Sun Sensors</u>														
Coarse	BBRC C-105	(1) *Inertial* Guidance	(1) *Inertial* Guidance	(1) *Inertial* Guidance			(1) *Inertial* Guidance	(1) *Inertial* Guidance	(1) *Inertial* Guidance	▲	▲		(1) Inertial Guidance	(1) Inertial Guidance
Fine	BBRC SS-107									▲	▲	▲		
Digital Aspect	Adcole Type 1402				▲(2)	▲								
<u>Earth Sensor</u>														
Low Altitude	TRW A-OGO				▲(2)									
High Altitude	TRW A-OGO					▲								
<u>Canopus Tracker</u>														
Medium Accuracy	Honeywell Mariner Mars '69									▲	▲			
High Accuracy	Kollsman KS 203-01											▲		
<u>Planetary Approach</u>														
High Accuracy	Kollsman Planet Tracker											▲		

(1) Optical sensors not required

(2) *Use is dependent upon duration of parking orbit*

the high-accuracy Canopus and planetary approach sensors are given in sec. 9 of vol. II, and are summarized in par. 7.2.3 which follows.

Earth-Synchronous Orbit Injection

The primary attitude reference will be the inertial elements of the guidance system during launch, injection into the parking orbit, coast-in-parking orbit, injection into the transfer orbit, and injection into synchronous orbit.

Launch and injection into the parking orbit will be accomplished using only the strapdown inertial guidance system. The duration of the parking orbit will vary between 15 min and 12 hr, depending upon the longitude desired for boost into the transfer orbit. If the duration of the parking orbit is long enough to require correction of the inertial reference system prior to boost into the transfer orbit, optical sightings will be utilized at this time. A low-altitude, earth-horizon sensor will be used to obtain a measurement of the vertical, in conjunction with sun sensors to obtain yaw alignment. Two choices of sun sensor configurations are apparent. Using a combination of coarse and fine sun sensors, vehicle maneuvers will be required to obtain a solar sighting, after which the vehicle will be returned to the earth-referenced attitude. Alternately, the use of a digital solar aspect sensor will permit a solar sighting to be obtained simultaneously with measurement of the vertical by the earth horizon sensor without requiring vehicle maneuvers. The latter choice is recommended.

After approximately five hours in the transfer orbit, correction of the inertial reference system alignment will again be required prior to injection into the earth-synchronous orbit. Again, the sun will be used as a reference for correcting the vehicle attitude in yaw, and the earth will be used as a reference for correction of the vertical. The same choice of sun sensors pertains, and the digital solar aspect sensor is again recommended.

Lunar Orbiter

As in the case of the earth-synchronous orbiter, the inertial elements of the guidance system will be utilized as the primary attitude reference for launch, injection into parking orbit, and for coast-in-parking orbit. The duration of the parking orbit may vary from 0 to 20 min. For this short-coast phase, no attitude update of the inertial system is required.*

After injection into the lunar transfer orbit, the primary attitude references will be the Sun and Canopus. The coarse and fine sun sensors will be used in conjunction with a single-axis Canopus tracker during this phase of the mission. The precision available from these sensors, together with the onboard inertial system and ground radio tracking aids (see subsec. 2.2), are adequate to perform the midcourse correction maneuver and deboost into lunar orbit without the use of additional electro-optical sensors for approach guidance.

Mars Orbiter

Injection into interplanetary transfer orbit will normally require parking orbit coasts not exceeding 30 min. Thus, no optical sensors are required during this phase of the mission.* After injection into the interplanetary transfer orbit, coarse and fine sun sensors will be used in conjunction with a Canopus tracker. However, the Canopus tracker is also used for the approach guidance to Mars. To achieve any significant improvement in the approach trajectory estimation over that available with earth-based tracking (subsec. 2.2), very high precision is required during this phase of the mission. Therefore, a Canopus sensor with higher tracking accuracy than that available in state-of-the-art equipment is required for this mission. A preliminary design concept for such a sensor is presented in sec. 9, vol. II, and is summarized in par. 7.2.3 following.

*The attitude error accumulated over short parking orbits of 20 to 30 min Ref. 7-4, duration is at most a few tenths of a degree. (See sec. 7, vol. II.) This error creates a lateral velocity error during the orbit injection burn approximately equal to the attitude error multiplied by the velocity accumulated during the burn. The resulting velocity errors (together with other accumulated errors) are well within reasonable correction capabilities for the midcourse maneuver.

Solar Probe with Jupiter Assist

Again, no electro-optical sensors are required prior to injection into the interplanetary transfer orbir. For the interplanetary cruise phase and for midcourse corrections, the sensors used will be identical to those used for the Mars mission. The use of a planetary approach sensor for approach to Jupiter is not required since no trajectory corrections are made subsequent to the midcourse maneuver.

7.2.2 Summary of Sensor Performance Characteristics

A summary of the error characteristics of the electro-optical sensors chosen to meet the requirements of the four missions is shown in Table 7-IV. The instrument error values in the table are derived in vol. II, sec. 9.

7.2.3 Summary of Sensor Design Characteristics

The design characteristics of the sensors selected for the various missions are described in detail in sec. 9, vol. II, and are summarized in the following paragraphs.

Whenever feasible, existing state-of-the-art equipment was selected. Some redesign is required in some cases to implement the interfaces with the digital computer in the manner required by the modular concept described in subsec. 6.4.

7.2.3.1 Sun Sensors

The candidate sun sensor system consists of a coarse sun sensor unit having a 4π sr total field-of-view and a fine sun sensor unit with a $2^{\circ} \times 2^{\circ}$ field-of-view. The coarse and fine sun sensors need not be mounted to a common reference surface. The coarse sun sensor assembly may be mounted outboard the spacecraft because it has sufficient environmental resistance, but the fine sun sensor assembly must be mounted inboard and integrated with the other fine optical sensors to achieve the required overall attitude reference accuracy.

TABLE 7-IV

SUMMARY OF ELECTRO-OPTICAL SENSOR ERRORS

Sensor	Type	Null Accuracy			Offset Accuracy (1)			
		Noise (rms)	Bias Instability	Fixed Bias	Offset	Noise (rms)	Bias Instability	Fixed Bias
Sun Sensors								
Coarse	BBRC C-105 Coarse Eyes (4)	N/A	N/A	N/A	±8°	Negligible	±35 min[2]	±7 min[2]
Fine	BBRC SS-107	Negligible	±1 min[2]	±21 sec[2]	±2°	Negligible	±1 min 39 sec[2]	±21 sec[2]
Digital Aspect	Adcole Type 1402	23 sec[2]	±2 min 32 sec[2]	±21 sec[2]	±32°	23 sec[2]	±2 min 32 sec[2]	±21 sec[2]
Earth Sensor								
Low Altitude and	TRW A-OGO	N/A	N/A	N/A	76° From NADIR	2 min 10 sec[2]	(2), (3), (6) ±10 min 20 sec	±1 min 24 sec[2]
High Altitude					8.7° From NADIR	2 min 10 sec[2]	(2), (3), (6) ±9 min 6 sec	±1 min 24 sec[2]
Canopus Trackers								
Midcourse	Honeywell Mariner Mars '69	30 sec[4]	±1 min[4]	±3 min[4]	±2.6°	30 sec[4]	±1.24 min[4]	±3 min[4]
Planetary Approach	Kollsman KS-203-01	8.4 sec[2]	±8 sec[2]	±5 sec[2]	±60°	8.4 sec[2]	±8 sec[2]	±5 sec[2]
Planet Tracker								
Planetary[5] Approach	Kollsman Planet Tracker	8 sec[2]	±14 sec[2]	-30±20 sec[2]	N/A	N/A	N/A	N/A

NOTES:
(1) All values are 1σ
(2) Composite error for two axes
(3) Earth oblateness and horizon altitude errors included in bias error
(4) Error on sensitive axis only
(5) Planet subtense = 2.8°
(6) Values correspond to one scanner assembly for both hi and low altitude, with precision positor calibration

a) Coarse Sun Sensor Description

The coarse sun sensor consists of four C-105 coarse-eye units and one switching eye, both developed by the Ball Brothers Research Corporation (BBRC). The coarse-eye units are mounted in back-to-back orthogonal pairs, as illustrated in Figure 7-3(a), to permit coverage of a full 4π sr field. The switching eye is aligned parallel to the roll axis and is mounted and aligned with the fine sun sensor.

Each coarse eye consists of a silicon solar cell covered by a transparent window of radiation-resistant glass, as detailed in Figure 7-3(b). For each axis, the electrical outputs of the coarse-eye pair are connected in opposing polarities across a low-resistance load. The voltage measured across the resistance is proportional to the sun elevation above the common plane of the coarse-eye pair, and hence, provides a one-axis error signal. This output characteristic is illustrated in Figure 7-3(c). The control system orients the spacecraft so as to null the output of each axis pair. Output polarity provides the control drive direction, eliminating the unstable null at 180°.

b) Fine Sun Sensor Description

The fine sun sensor is the BBRC SS-107 fine sun sensor assembly. Four BBRC F-125 wide-angle fine eyes are used in an orthogonal configuration, as indicated in Figure 7-4(a), to provide error signals in two orthogonal axes over a $\pm 2^\circ$ linear range. The fine sun sensor relies on the coarse sun sensor to provide coarse orientation until the sun is within $\pm 6^\circ$ of null.

Each of the fine eyes consists of an objective lens, a knife-edge reticle, a filter, and a silicon solar cell. The output current from each solar cell is linearly proportional to the displacement of the sun angle from the optical axis, as indicated in Figure 7-4(b). A pair of sensors, oriented 180° apart and with their outputs differenced, provides a total linear error-sensing capability of $\pm 5^\circ$ from the null plane; however, the electronics are designed to saturate at $\pm 2^\circ$, the maximum range required. Together with a second pair of sensors, orthogonal to the first pair, the yaw and pitch angular offsets are completely defined over a $2^\circ \times 2^\circ$ range.

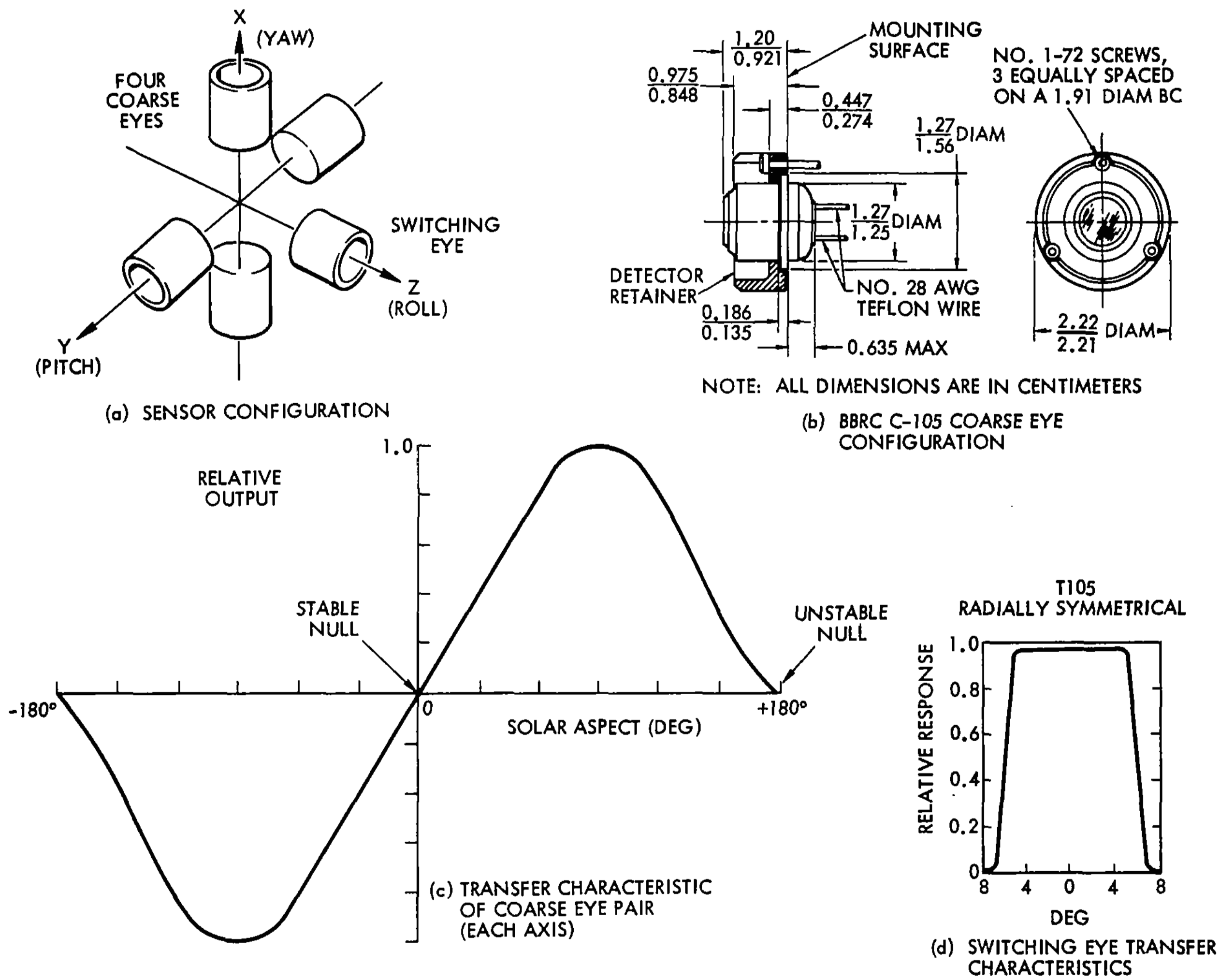

Figure 7-3. Coarse Sun Sensor Assembly Characteristics

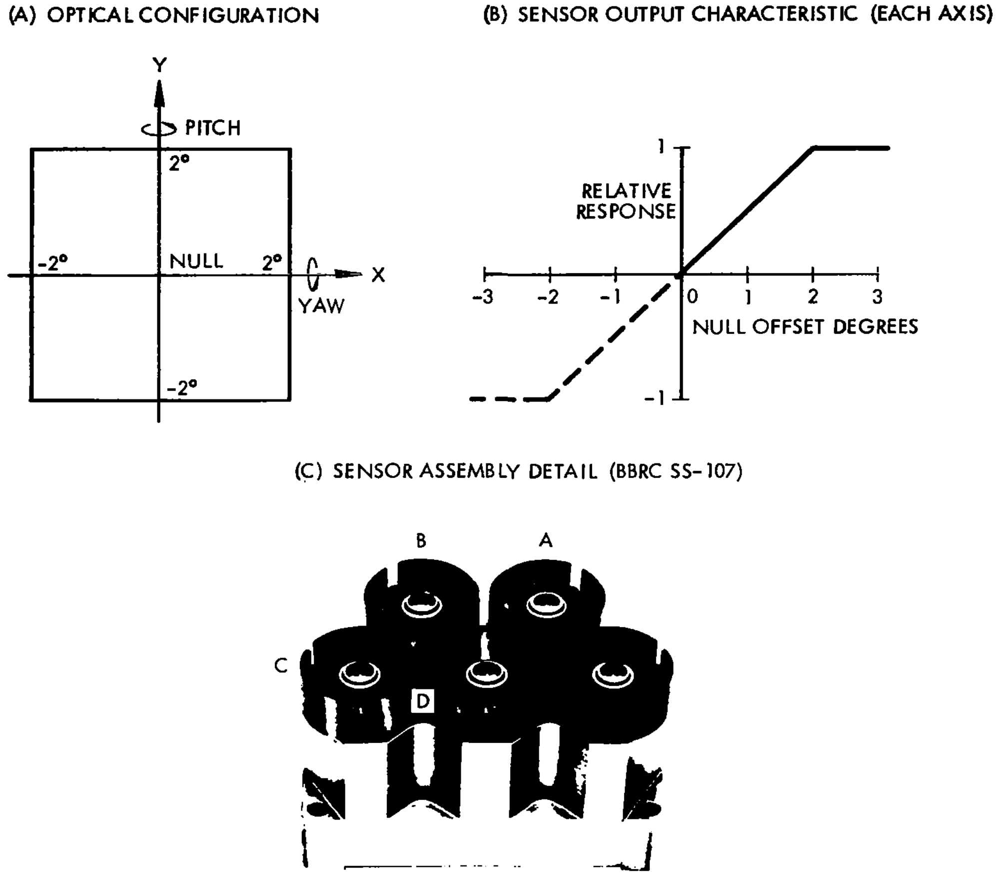

Figure 7-4. Fine Sun Sensor Assembly

The fine-eye pairs are mounted in a block as shown in Figure 7-4(c) and aligned to provide a nulled electrical output when the control axes (yaw and pitch) are normal to a referenced plane on the rear of the block. This reference plane is defined by three lapped pads on the mounting surface of the block. The control planes of the yaw and pitch axes are aligned normal to each other and are referenced to a machined surface on one side of the block.

The fine-eye assembly is very simple, mechanically rigid, and electrically stable. The performance specification indicates an accuracy better than 2 arc min (3σ) over a temperature range of -20° to $+85^{\circ}C$. The specifications for the course and fine sun sensors are summarized in Table 7-V.

TABLE 7-V

COARSE AND FINE SUN SENSOR SPECIFICATIONS

Fine Sensor Assembly*	
Accuracy (fine eye pair at null)	±2 arc min, 3σ
Peak output (short-circuit current in direct sunlight)	1.5 ma nominal
Angular range (fine eye pairs)	$\pm 15^{\circ}$ nominal
Angular sensitivity (front edge)	5 μamp/arc min
Temperature operating range	$-20^{\circ}C$ to $+85^{\circ}C$
Coarse Sensor Assembly	
Field of view	4π ster
Null accuracy (each axis)	$\pm 1^{\circ}$
Linearity (over $\pm 20^{\circ}$ each axis)	±10%
Physical Characteristics (Includes Electronics)	
Size	700 cm^3
Weight	0.4 kg
Power	500 mW

*Manufacturer's specifications.

c) Solar Aspect Sensor

The digital aspect sensor recommended for the earth-synchronous orbit mission is a device designed and manufacturered by the Adcole Corporation. This sensor measures two orthogonal components of the sun's offset from the instrument reference axis and presents the data in digital form. The performance characteristics of this unit are summarized in Table 7-VI. The operation of the sensor is described in subsec. 9.4, vol. II.

TABLE 7-VI

DIGITAL SOLAR ASPECT SENSOR SPECIFICATIONS*

Model	Adcole type 1402
Field-of-view	$64^{\circ} \times 64^{\circ}$
Resolution	$1/64^{\circ}$
Accuracy	2 arc min
Output	Two 12-bit words
Operating temperature range	-70 to 100°C
Size	1.3 x 1.3 x 2.1 cm
Weight	0.15 kg
Power	None required

*Manufacturer's specifications

7.2.3.2 Earth Sensor (Horizon Scanner)

In the parking orbit of the earth-synchronous mission, the half-angle subtended by the earth is approximately 75°; at synchronous altitude this half-angle is 8.7°. Although the highest accuracy would be obtained by using one earth sensor assembly at the low altitude of the parking orbit and a second earth sensor assembly at synchronous altitude, it was determined in this study that a common earth sensor assembly could be used at both altitudes and still provide sufficient accuracy to accomplish the prescribed missions.

The advanced OGO horizon tracker, developed by the Advanced Techbology Division of American Standard (Figure 7-5), is the earth sensor recommended. The instrument consists of four sensors arranged at 90° intervals in yaw, utilizing linear scanning and edge tracking of the horizon as shown in Figure 7-6. The method of interconnection of the four sensors (one is redundant) is shown in Figure 7-7. The electronic configuration for a single channel is shown in Figure 7-8. Performance requirements for the instrument and other pertinent operating characteristics are shown in Table 7-VII. The operating principles are described in detail in subsec. 9.5, vol. II.

The selection of this instrument was based on the following considerations.

a) For use in a nonspinning vehicle, edge-tracking, radiance-balance, horizon-sector, and conical scanners may be considered. The radiance balance technique was rejected due to low accuracy. The latter two were rejected from the standpoint of reliability, as rotating mechanisms are required for scan generation.

b) The edge-tracking sensor has the advantage of having a scanning mechanism which utilizes flexural pivolts of high reliability.

c) The edge-tracking technique inherently has a higher signal-to-noise ratio than the conical scanning method.

d) The spectral bandpass utilizing the 14- to 16-μ CO_2 absorption band provides improved definition of the infrared horizon of the earth in comparison to previous sensors utilizing infrared wavelengths shorter than 14 μ, in which inaccuracies resulted due to discontinuities in the infrared horizon.

7.2.3.3 Canopus Trackers for Lunar Orbiter and Interplanetary Missions

In considering the selection of the Canopus trackers for the specified missions, three applications were taken into account. These are the Lunar Orbiter mission of only a few days duration; the interplanetary mission of 7-1/2 to 15 months duration; and the Mars approach guidance phase of only a few hours duration. A summary of comtemporary Canopus trackers applicable to these missions is contained in Table 7-VIII.

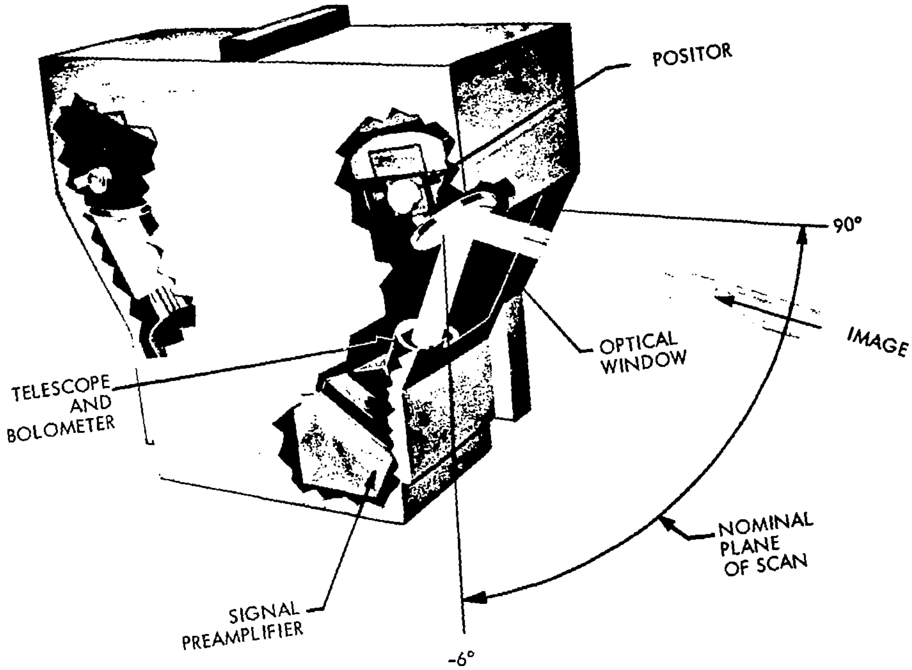

Figure 7-5. A-OGO Horizon Sensor

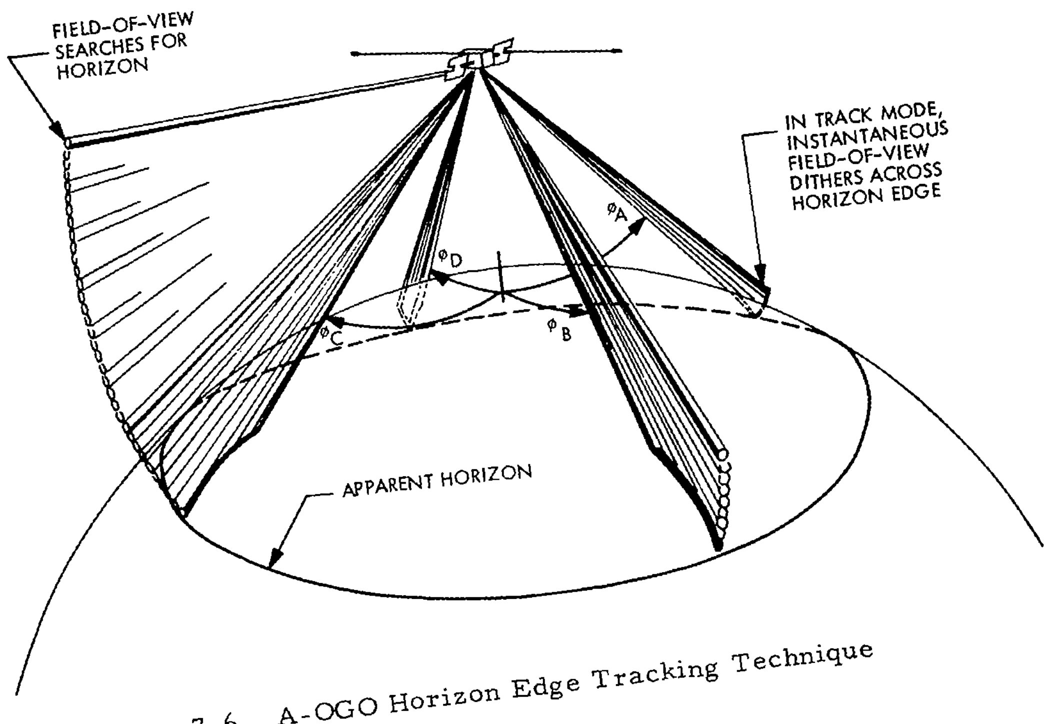

Figure 7-6. A-OGO Horizon Edge Tracking Technique

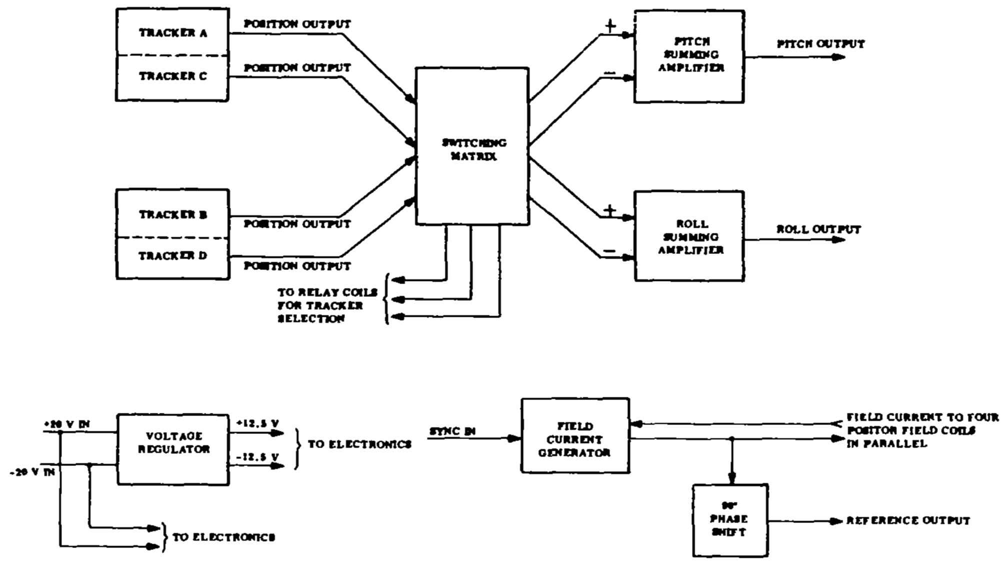

Figure 7-7. A-AGO Horizon Sensor System Block Diagram

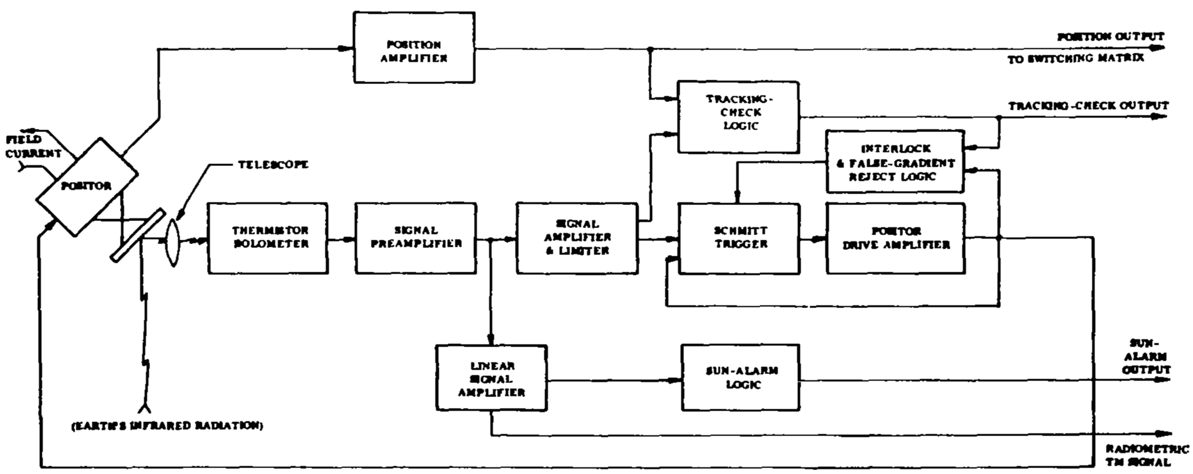

Figure 7-8. A-AGO Tracker Block Diagram (Single Channel)

TABLE 7-VII

A-OGO HORIZON SENSOR SYSTEM SPECIFICATIONS*

Optical Characteristics	
a) IR detector	Immersed thermistor bolometer
b) IR spectral bandpass	14.0 to 16.0 μ
c) Telescope field-of-view	1.2 deg at half-response contour
Sensor Outputs	
a) Pitch/roll	2461-cps signal with amplitude proportional to roll and/or pitch attitude error
Physical Characteristics	
a) Size	5000 cm^3 (maximum)
b) Weight	7.6 kg (including electronics)
c) Power required	10 w (nominal), 12 w (maximum)
Performance	
a) Tracking range (each of 4 trackers)	-2 to +85 deg (min)
b) Tracking rate	>15 deg/sec
c) Operational range	±30 deg (±45 deg from nominal)
d) Altitude range	220 to 150,000 km or 90 to 110,000 km
e) Accuracy**	≤0.05 deg (3σ)
Null	
±10 deg***	≤0.10 deg (bias) + 0.05 deg (3σ)
f) Reliability (for 3 or 4 trackers operating)	0.95 for 1 year (present parts) 0.98 for 1 year (highest reliability parts available)
g) Operational life	≥1 year
h) Storage life	3 years
i) Pitch and roll scale factors	0.4 v rms/deg
j) Position-output scale factor	0.1 v rms/deg
k) Noise	±0.02 deg peak-to-peak at 0.6 Hz bandwidth
Environmental Levels	
See Section 5.0, Volume II	

* Manufacturer's specifications.

** Excluding geometric cross-coupling errors (which can be calibrated out) and errors due to horizon anomalies and earth oblateness.

*** Simultaneous roll and pitch.

TABLE 7-VIII

SUMMARY OF CANOPUS TRACKERS

Application	Lunar Orbit	Interplanetary Midcourse Guidance		Mars Approach Guidance
Item	Lunar Orbiter Canopus Sensor	Mariner IV Canopus Sensor	Mariner Mars 69 Canopus Tracker	KS 203-01 Star Tracker
1. Manufacturer	ITT Federal Laboratories	NASA-JPL/Barnes Engineering Company	Honeywell Radiation Center	Kollsman Instrument Company
2. Application	Lunar Orbiter	Mariner IV	Mariner '69	Classified
3. Optical system	Refractive — 7 elements + corrector 20 mm f/1.0	Semi-solid Cassegrain Schmidt 20 mm fl f/0.6 T/1.0	Refractive — Hypersil	Cassegrain/Mangin mirror 80 mm f l f/1.25
4. Detector	ITT Type FW 143 photomultiplier	CBS Type CL 1147 image dissector	CBS Type CL 1147 image dissector	Quadrant silicon cell
Spectral response	S-20	S-11	S-11	0.6 μ to 1.1 μ
Aperture dimensions	0.024 x 0.435 cm	0.3 x 4.0 mm	Not spec	N/A
5. Instrument field-of-view				
Roll (total)	8.2° (±6° after acquisition)	4°	9°	1°
Pitch (total)	18°	30°	35.8°	1°
Instantaneous	1° (roll) x 18° (cone)	0.85° (roll) x 11° (cone)	1.05° (roll) x 11° (cone)	1° x 1°
6. Gimbaling	Mechanical adjustment for launch window	Electronic	Electronic ±3° (roll) ±17.9° (cone)	Mechanical
7. Scanning				
Roll (search)	±4.1° triangular at 14 Hz	±2° sinusoidal at 1 KHz	±1° sinsoidal at 1.2 KHz	100-Hz nutation
Roll (track)	±1.5° triangular at 800 Hz + dc bias	±2° sinusoidal at 1 KHz	±1° sinusoidal at 1.2 KHz	100-Hz nutation
Pitch	None	5 programmed, 3 optional increments (4.6° ea)	5 programmed increments (6.2° ea)	100-Hz nutation + incremental search
8. Stellar sensitivity	-1.92 to + 0.08 m	-2.1 to +0.6 mag (16/1)	0.04 to 3.0 x Canopus mag (75/1)	100 stars to +1.8 silicon mag
9. Linear range	±6°	±0.85°	±2.6°	5 x 5 min
10. Electronic bandwidth	0.75 Hz	0.312 Hz	Not spec	6.9 Hz (acquisition)/ 1.8 Hz (track)
11. Time constant	0.2 sec (roll axis)	0.5 sec	0.5 sec (max)	0.1 sec (star presence)/ 10 sec (lock-on)
12. Acquisition rate	Not spec	0.116°/sec	Not spec	0.6°/sec (max)

TABLE 7-VIII

SUMMARY OF CANOPUS TRACKERS (Continued)

Application	Lunar Orbit	Interplanetary Midcourse Guidance		Mars Approach Guidance
Item	Lunar Orbiter Canopus Sensor	Mariner IV Canopus Sensor	Mariner Mars 69 Canopus Tracker	KS 203-01 Star Tracker
13. Accuracy				
• At null				
noise	15 arc sec rms	0.114° (6.85 arc min) pp (Note 1)	30 sec rms (1σ) (Note 1)	8 sec rms (1σ) (Note 1)
bias	50 arc sec (stability)	0.1° (6 arc min)	+1.0 min (1σ)	±8 sec (1σ)
alignment	Not spec	Not spec	±3 min (roll)/±6 min (cone)	±5 sec
• Off-axis				
noise	Not spec	Not spec	30 sec rms (1σ)	8 sec rms (1σ)
bias	Not spec	Not spec	±1.24 min (1σ)	±8 sec (1σ)
alignment	Not spec	Not spec	±3 min (roll)/±6 min (cone)	±5 sec
16. Major error source	I. D. and aperture alignment and edge irregularities	Deflection nonlinearity, mech alignment	Deflection nonlinearity, mech alignment	Gimbals and encoders
17. Sun protection	CdS sensor — 100 ft-c threshold	CdS sensor/1000 ft-c threshold	1000 ft-c threshold	Not spec
18. Weight	3.2 kg (7 lb)	2.3 kg	0.363 kg	13.7 kg
19. Volume	10.2 x 14.0 x 30.5 cm (4 x 5.5 x 12 in.)	10.2 x 12.7 x 28.0 cm	30 x 13 x 11 cm (tracker) 31 x 19 x 13.5 cm (baffle)	1.0 ft^3
20. Power	3.1 W at 21 Vdc, 4.95 W at 31 Vdc	1.5 W (av)	5.5 W plus 6.5 W for sun shutter	24 W
21. MTBF	Not spec	Not spec	Not spec	16,000 hr
22. Primary improvements	N/A	N/A	Glare baffling, larger field-of-view, ruggedization, wider photometric range	N/A

Note 1: Accuracy figures are specified for the sensitive axis only.

a) Canopus Tracker for Lunar Orbit

Two Canopus trackers have been space proven in Lunar missions. The first is the instrument developed by ITT Federal Laboratories for use in the Lunar Orbiter program. The second is the instrument developed by the Hughes Aircraft Company, Santa Barbara Research Center (SBRC), for the Surveyor program. A detailed description for the latter instrument is contained in Ref. 7-1, vol. III.

Although both of these instruments havw been proven in lunar flight, the Mariner IV and Mariner Mars 69 interplanetary Canopus trackers may also be considered for the lunar mission. The Mariner Mars 69 instrument is preferred because it will incorporate a number of improvements over the Mariner IV design. These improvements are: better glare baffles; increased cone angle range; ruggedization; and a wider photometric acceptance range.

The Mariner Mars 69 Canopus tracker has been selected in this study to fulfill the requirements of both the lunar and interplanetary missions. Employing the same type of tracker for both missions will simplify the configuration of the proposed guidance system, eliminating duplication of interfaces.

The selection of the Mariner Mars Canopus is based upon the following advantages over the ITT Federal Laboratories Lunar Orbiter Canopus tracker:

1) Improved optical baffling to reduce spurious signals induced by solar glare

2) A larger photometric acceptance range of stellar irradiance (75/1, rather than 6.25/1)

3) A total cone angle variation range of 35.8°, rather than 18°

4) Comparable accuracy

5) Caoparable weight

6) A smaller power requirement (except during periods of sun shutter operation).

The Mariner Mars 69 Canopus tracker is also preferred over the SBRC Canopus tracker used on the Surveyor program because the latter uses several mechanisms which are objectionable from the standpoint of long lifetime, i.e., a single-axis cam-driven scanning mirror and a rotating reticle for modulation of the incident star radiation. A block diagram of the Canopus tracker is shown in Figure 7-9.

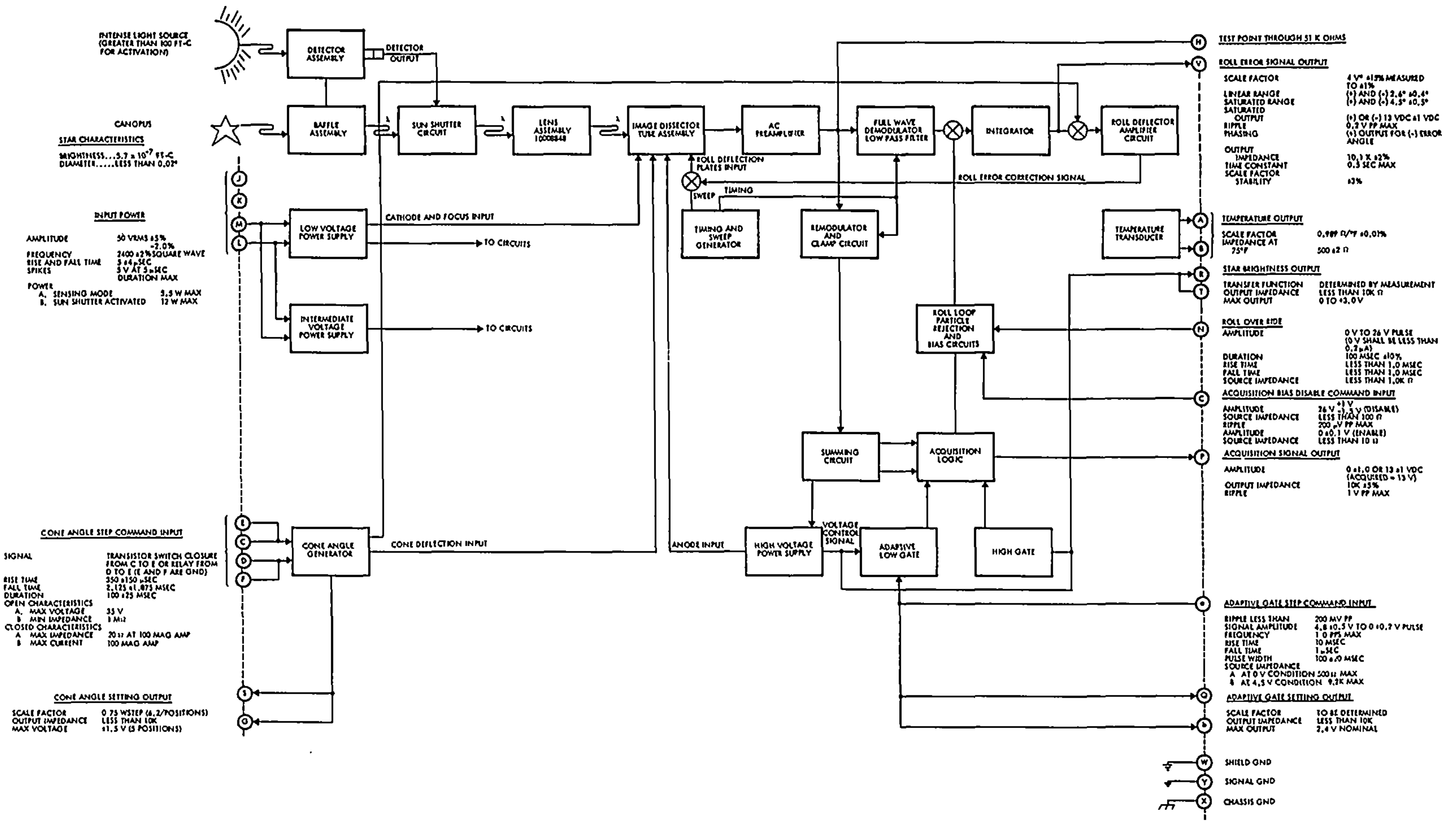

Figure 7-9 Block Diagram of Canopus Sensor for Lunar Interplanetary Attitude Control

b) Canopus Tracker for Interplanetary Midcourse Attitude Control

The only tracker that has been spaced proven in interplanetary midcourse guidance is the Mariner IV Canopus sensor. However, as the Mariner Mars 69 equipment represents an improved version of the Mariner IV design, the former is selected for this mission. The current status of this equipment is that all flight-qualified units and spares recently were delivered to NASA-JPL by the Honeywell Radiation Center.

c) Canopus Tracker for Mars Approach Guidance

For the Mars approach guidance phase, an instrument of very high accuracy is required. Computer simulation of this mission showed that to effect any significant improvement over doppler-only orbit determination, the composite accuracy due to variable bias of the fine sun sensor, Canopus tracker, and Mars approach sensor must be approximately one arc minute. This composite error value requires the use of a Canopus tracker with a total variable bias error in the order of ±15 arc sec. The Mariner Mars 69 Canopus tracker, which has been selected for midcourse guidance, cannot meet this requirement; therfore, a precision gimbaled star tracker, the Kollsman type SA 201-03 Canopus tracker, was selected. Although this specific configuration has not been developed yet, the proposed design is based upon that of the KS-197, currently in development for a military space program. The estimated accuracy of this tracker used in a single-axis control mode is 12.5 arc sec rss (1σ), including all errors due to angular equivalent noise, variable bias, and alignment. Based upon experience in the development of the KS-197 equipment, Kollsman proposes a development schedule of one year, which can be reduced to nine months if dictated by program schedule requirements. TRW Systems believes that this schedule can be quoted with a 100% confidence level, based upon the Kollsman Instrument Corporation's current experience in developing similar equipment. A block diagram of this Canopus tracker is shown in Figure 7-10.

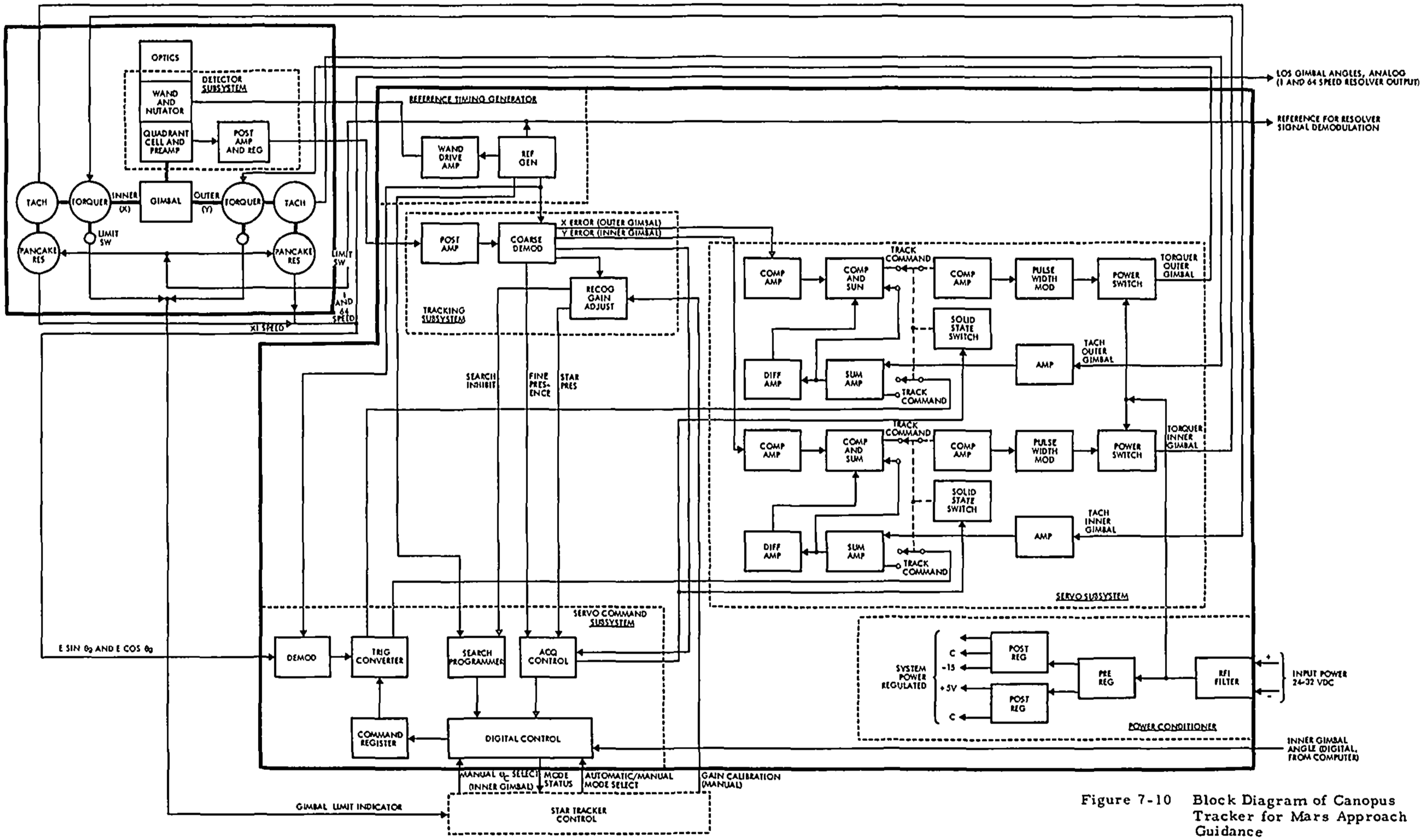

Figure 7-10 Block Diagram of Canopus Tracker for Mars Approach Guidance

7.2.3.4 Planet-Approach Sensor

The instrument selected for use as a planet-approach sensor is currently being developed by the Kollsman Instrument Corporation for the NASA Electronics Research Center. Selection of this instrument for this application is based upon: 1) the results of the survey conducted during Task II of the current study contract, 2) the anticipated performance of the proposed instrument, and 3) maturity of developmental status.

Computer simulation of the approach guidance phase of the Mars mission determined that the composite error due to variable bias of the fine sun sensor, precision Canopus tracker, and planetary approach sensor must be in the order of 1 arc min for the optically aided inertial guidance to improve significantly the approach position and velocity knowledge gained by ground-based doppler tracking. This composite error value requires a planet-approach sensor with a total error due to variable bias of about ±15 arc sec.

Referring to the survey conducted during Task II of this study (Ref. 7-1, vol. III), four planet-approach sensors were identified. The first, developed by the Barnes Engineering Company for the NASA Jet Propulsion Laboratories for use in the Mariner program, employs a thermistor bolometer detector with mechanical scanning to produce a rosette search and track pattern. However, the accuracy of this instrument, ±90 arc sec, is inadequate for this application.

The second instrument, the Barnes lunar and planetary horizon sensor, also developed for the NASA Jet Propulsion Laboratories, uses four thermopile arrays, electronically scanned. Again, the quoted accuracy of (±0.5° is not adequate for the requirements of this study.

A third instrument is under development by the Lockheed Missile and Space Company for the NASA Ames Research Center. Using an image dissector tube, the design objective is an accuracy of ±1.6 arc sec. However, this accuracy can also be obtained by using the very long focal length of the IR/OAO primary telescope. The planet tracker is not self-contained; it utilizes a portion of the irradiance collected by the primary telescope. If an optical system of considerably shorter focal length were used, the accuracy of the subsystem would decrease accordingly. This instrument is not considered applicable to this study because development of a compact optical system would be required in addition to a mechanism for pointing the line-of-sight for initial search and acquisition.

The fourth instrument included in the above survey is the planet horizon sensor developed by Northrop-Nortronics for the NASA George C. Marshall Space Flight Center. This instrument uses four thermopile arrays in a cruciform configuration, electronically scanned, similar to the Barnes lunar and planetary horizon sensor. However, only one instrument was developed in early experimental form, and the quoted angular accuracy of ±6 arc min is not adequate for the requirement of this study.

A fifth instrument, not included in the above survey, was recently under development by Electro-Optical Systems, Inc., for the NASA Jet Propulsion Laboratories. This instrument uses an image dissector tube, sensing planetary radiance in the near-visual portion of the spectrum. Pointing was accomplished by using mechanically driven contra-rotating optical wedges. This instrument was only partially completed, because of the deletion of the proposed approach guidance experiment from the Mariner 69 program. Selection of the Kollsman instrument over the Electro-Optical Systems instrument was based primarily upon maturity of developmental status.

Since the completion of the survey conducted in Task II of this contract, development of the planet tracker has been underway at the Kollsman Instrument Corporation under contract to the NASA Electronics Research Center. This is a gimbaled instrument using an image dissector tube as a detector. Precision gimbals and angle encoders are used, similar to those in the KS-197 star tracker (see Figure 7-11 for block diagram). This instrument, selected for the Mars approach guidance application in the use of a narrow optical field-of-view in conjunction with precision gimbals and angle encoders, offers a considerable improvement in accuracy over a strapdown instrument using a wide optical field-of-view. The rationale for this stagement is as follows.

The primary error sources in a detection system utilizing an image dissector tube are nonlinearities in the deflection coils, alignment of the image dissector aperture with respect to the optical line-of-sight, and edge irregularities in the image dissector aperture. Used with an

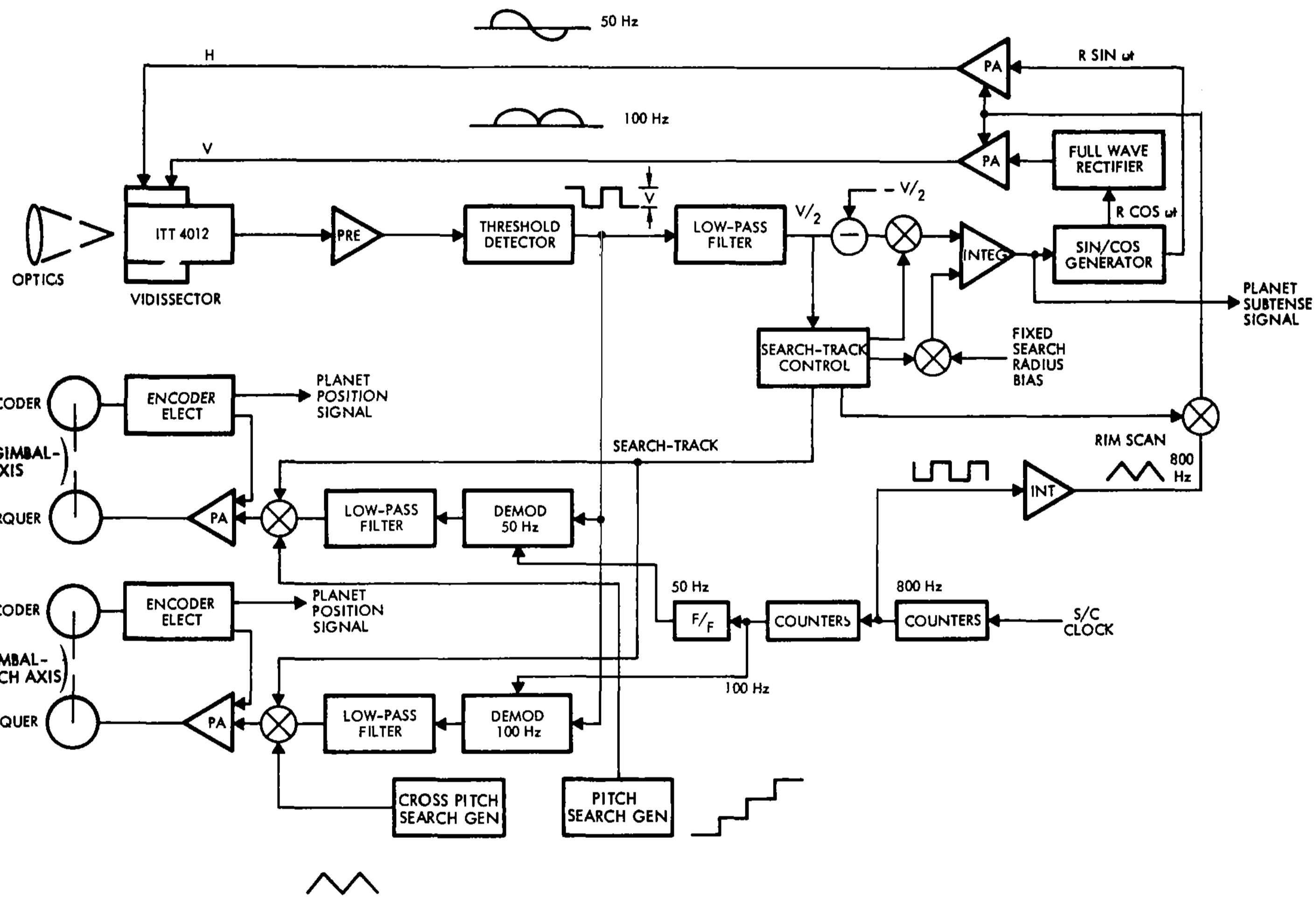

Figure 7-11. Planet Approach Sensor Functional Block Diagram

optical system of a given focal length, the magnitudes of the above errors, in conjunction with the dimension of the focal length, define the angular accuracy of the instrument. Wide field-of-view systems inherently require the use of optical systems with short focal lengths, resulting in low accuracy. Conversely, narrow field-of-view systems can utilize optical systems with long focal lengths, resulting in high-angular accuracy. Consequently, the angular accuracy of the instruments is increased using an optical system with a long focal length and a narrow field-of-view.

Pointing over a wide angular range can be accomplished by the use of precision gimbals and angle encoders. With precision machining, gimbals can be produced with only a few arc seconds of angular error over the complete pointing range. The limit of accuracy normally is determined by the angle encoders, and, in a more practical sense, by the degree to which encoding is required (number of counts per revolution) and the complexity of the associated encoding electronics.

The composite error anticipated in the Kollsman planet tracker caused by variable bias on both axes is estimated to be 16.7 arc sec (1σ). This error value is based upon data presented in Table 7-IX.

The optical field-of-view and pointing angle requirements for the planet-approach sensor were determined for both the Type I and Type II trajectories used in this study (Figure 7-12). Using the Type I trajectory with a value of v_∞ of 3.09 km/sec, and assuming that the planet approach sensor will be used over a nominal range from 10 days to 1/2 day prior to encounter, the variation in both cone and clock angle will be approximately five degrees. The apparent diameters of the planet increases from a value of 0.15° to 2.82°. Using the Type II trajectory with a value of v_∞ of 2.8 km/sec, a cone angle variation of approximately five degrees is required again, but the clock angle is nearly invarant. The apparent diameter of the planet increases from a value of 0.16° to 3.22°.

TABLE 7-IX

SUMMARY OF PLANET APPROACH SENSOR ERRORS
(OPTICAL FIELD-OF-VIEW = $4^{o} \times 4^{o}$. ALL VALUES IN ARC SEC, 1σ)

Planet subtense*	2.82°						0.15°					
	Noise		Variable Bias		Fixed Bias		Noise		Variable Bias		Fixed Bias	
	Pitch	Cross Pitch	Pitch	Cross Pitch	Pitch	Cross Pitch	Pitch	Cross Pitch	Pitch	Cross Pitch	Pitch	Cross Pitch
1. Deflection coils												
Linearity (0.5%)					0	-16.0					0	-0.8
Symmetry (0.1%)					±3.2	±3.2					±0.2	±0.2
Variation (0.1%)			±3.2	±3.2					±0.2	±0.2		
2. Vidisector abberation (Mfg. spec = 1.27×10^{-3} cm max)					±10.6	±10.6					±0.6	±0.6
3. Optical distortion (0.05%)					±2.5	±2.5					±0.1	±0.1
4. Planet radiance variation 4:1			±5.6	±5.6					±5.6	±5.6		
5. Planet oblateness					±6.4	-14.2					±0.3	-0.8
6. Electronics												
Threshold noise	Negligible	Negligible					Negligible	Negligible				
Drift-sweep gen. (±5 mV)			±3.6	±3.6					±3.6	±3.6		
Drift sine cosine gen. (±5 mV)			±3.6	±3.6					±3.6	±. 6		
7. Mechanical alignment head assembly					±5.0	±5.0					±5.0	±5.0
8. Thermal drift-head assembly			±2.0	±2.0					±2.0	±2.0		
9. Gimbals												
Resolver digitizer	±5.7	±5.7					±5.7	±5.7				
Mechanical error			±4.9	±5.8					±4.9	±5.8		
Mechanical stress			±1.6	±1.6					±1.6	±1.6		
Servo	±1.3	±1.3					±1.3	±1.3				
10. Installation												
Mechanical alignment					±5.0	±5.0					±5.0	±5.0
11. RSS per axis	±5.9	±5.9	±9.9	±10.4	±14.8	-30.2±13.4	±5.9	±5.9	±9.2	±9.9	±7.1	-1.6±7.1
12. RSS both axes	8.3 RSS		14.4 RSS		-30.2±19.6 RSS		8.3 RSS		13.6 RSS		-1.6±10.0 RSS	

*Approach velocity = 3.09 km/sec

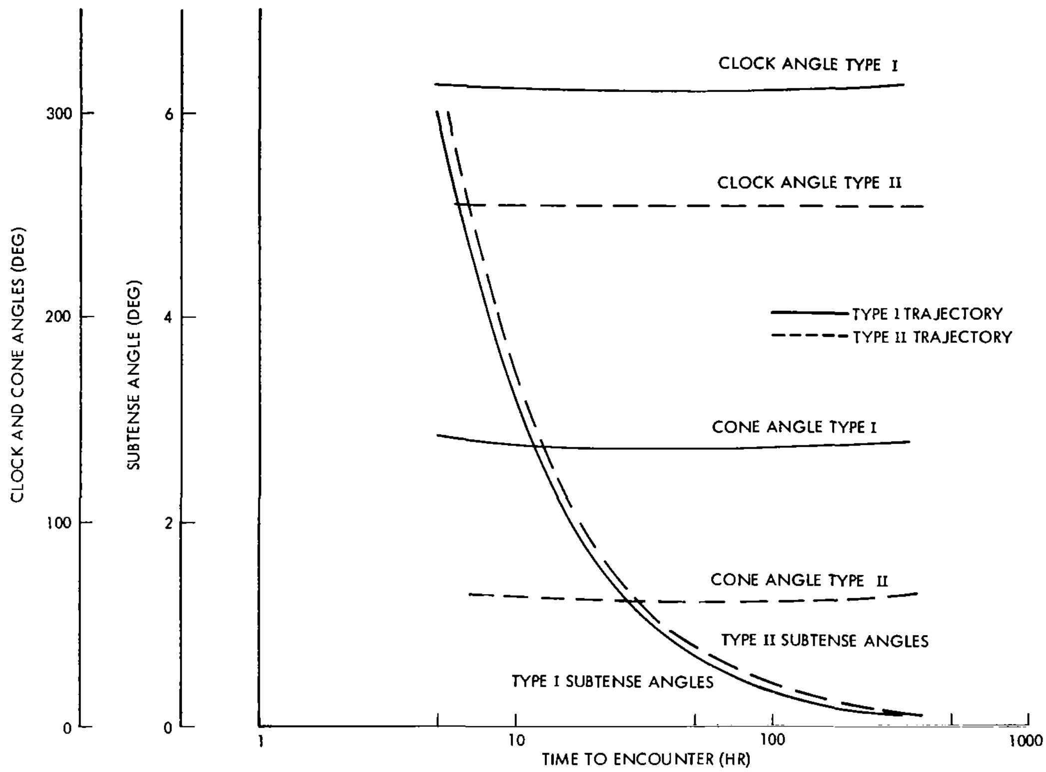

Figure 7-12. Clock and Cone Angles and Apparent Planet Angular Subtense versus Time to Encounter

Using the above values as nominal, an optical field-of-view of $4^{\circ} \times 4^{\circ}$ was selected to accommodate the maximum value of apparent planet diameter. To permit reasonable variations in vehicle attitude and flexibility in trajectory selection, a search field pattern of $20^{\circ} \times 20^{\circ}$ square was selected ($\pm 10^{\circ}$ on both gimbal axes).

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7-1. "Radio/Optical/Strapdown Inertial Guidance Study for Advanced Kick Stage Applications," Final Report, TRW Document Number 07398-6017-R000, 30 June 1967.

7-2. "Standardized Space Guidance System Final Report, Phase 1a," TRW Space Technology Laborarоties Report SSD-TDR-64-132, Volume III, 29 May 1964.